CR-112249

A STRUCTURAL DESIGN FOR AN EXTERNALLY BLOWN FLAP (EBF) MEDIUM STOL RESEARCH AIRCRAFT

BY THE

ADVANCED TRANSPORT TECHNOLOGY ENGINEERING STAFF

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HAMPTON TECHNICAL CENTER

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FOREWARD

The work described herein was conducted by the Hampton Technical Center of LTV Aerospace Corporation, under NASA Advanced Transport Technology Project Manager, Mr. W. J. Alford, and Technical Monitor, Mr. T. F. Bonner, Systems Engineering Division, NASA Langley Research Center. The report was prepared by the Advanced Transport Technology Engineering Staff under the direction of Mr. R. R. Lynch, the Hampton Technical Center Advanced Aircraft Technology Manager.

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SUMMARY

Recent studies have indicated certain principle high-lift systems that appear attractive for application to STOL aircraft. One of these is the externally blown flap (EBF) concept where engine air is directed over the wing and flap. In the design phase, an understanding of the dynamic characteristic of an externally blown flap high-lift wing is required. In order to generate a more thorough data base, a computer program to predict, by reference to structural drawings, the dynamic response of a high-lift STOL wing appears essential.

The primary objective of this report is to provide structural stiffness, weight and loads information to L. R. C. for input into a dynamic model analysis computer program. This data is presented in the form of sketches, weight and dynamic loads information graphs and tables for an external blown, triple-slotted flap, high-lift STOL transport wing.

The design of a full cantilever wing for an external blown flap (EBF) experimental STOL research aircraft was developed to the detail of locating major components of the wing such as engine locations, leading and trailing flap panel trim, and spoiler and aileron locations. Major load points were determined and primary structural load paths developed. The functional and structural design studies of the major components were investigated to assure feasibility and to permit structural analysis.

The structural analysis of the wing box and component parts was conducted at a preliminary design level. "Smear" analysis method was used to compute total cover thickness of wing bending material and arbitrary assumptions of allowable stress and percent effective material were applied to account for combined stresses and fatigue considerations. The flap tracks and support structure was sized at critical points with the flap in the extended position.

The engine pylon is a cantilever beam extending forward of the wing and supporting the concentrated load of the engine. Due to the critical nature of its dynamic response, a more detailed analysis is presented for the engine pylon. The wing was analyzed for nacelle total weights, exclusive of the mounts, with the nacelles both rigidly and elastically mounted.

Weight, mass distribution, and moment of inertia data is summarized in table form and presented pictorially by drawing layout. Weight data was obtained by three methods:

1. Actual know weight of components.

2. Determined from preliminary stress sizing.

3. Statistical weight estimating methods.

INTRODUCTION

NASA is engaged in a concerted effort to provide a firm technology foundation for the design, development, fabrication, and operation of safe, reliable, quiet, and economical fan-jet STOL transport. One phase, design, is concerned with the dynamic flutter characteristics of an external blown flap high-lift wing.

In order to generate a more thorough data base required by designers, it is necessary to establish a computer program to predict by reference to structural drawings, the dynamic response of a STOL wing.

The primary objective of this report is to supply structural stiffness, weight, and loads information to L. R. C. for a dynamic model analysis computer program. To meet this objective, drawings have been developed in sufficient detail to permit stress, dynamic loads and weight information graphs and tables to be prepared for an external blown, triple-slotted flap, high-lift STOL transport wing.

SYMBOLS

A	Area, in^2
Ae	Area enclosed, in^2
Al	Area lower, in^2
Au	Area upper, in^2
B	Box width, in
b	Wing span, feet
C.G.	Center of Gravity
CnCq	Section normal force, lbs/in
CmC^2q	Section hinge moment, in-lbs/in
c	Distance from neutral axis, in
E	Modulus of elasticity, lbs/in^2
EI	Bending stiffness, in^2-lbs
F	Force, lbs
Fs_{CR}	Shear buckling stress, lbs/in^2
f	Stress, lbs/in^2
G	Load factor or modulus of rigidity, lbs/in^2 depending on use
GA	Shear stiffness, lbs
GJ	Torsional stiffness, in^2-lbs
h	Height, in
hz	Hertz, cycles per second
I	Area moment of inertia, in^4
Izo, Iyo, Izo	Mass moment of inertia, lbs-in about respective axis
I.D.	Inside diameter

$J = \sqrt{EI/P}$	Beam-column coefficient, in
k	Strouhal factor, for unsteady aerodynamics
Ks	Shear buckling constant
L	Length, in
M	Bending moment, in-lbs
N	Load, lbs/in
N.S.	Normal station, in
O.D.	Outside diameter
P	Load, lbs
PSI	Stress, lbs/in^2
q	Shear flow, lbs/in or dynamic pressure, lbs/in^2 depending on use
R	Reaction load, lbs
S	Side load, lbs
T	Torque, in-lbs
Tc	Engine thrust, cruise, lbs
Tm	Engine thrust, max, lbs
t	Thickness, in
$\bar{t}l$	Cover thickness lower (smeared), in
tu	Cover thickness upper (smeared), in
$t_{sl} = AL/2B$	Thickness of lower skin, in
$t_{su} = Au/2B$	Thickness of upper skin, in
V	Vertical shear or load, lbs - Velocity in knots in Dynamic Loads
W	Weight, lbs
w	Uniform load, lbs/in
$\bar{x}$	Distance of mass from x reference axis, in
Y	Distance from centerline along wing, feet
y	Deflection, in
$\bar{y}$	Distance of mass from y reference axis, in

α	Angle of attack, degrees
η, eta	Fraction of wing semi-span
xi	Modal amplitude
ρ = I/A	Radius of gyration, in
ω	Circular frequency, radians/sec

SUBSCRIPTS

avg	Average
e	Effective
F	Critical speed, flutter speed
ult	Ultimate
tot	Total
w	Web
wb	Wing bending
x,y,z	Rectangular Cortesian coordinates
N	Normal - Nacelle, in Dynamic Loads
H	Horizontal
V	Vertical

MATHEMATICAL CONVENTIONS

Σ	Sum
=	Equal
+	Plus
-	Minus
x	Multiply by
$\sqrt{\ }$	Square root
$\pm$	Plus or minus

Section I. STRUCTURE

Wing

The wing for the experimental STOL transport research airplane has a 17% thickness supercritical airfoil. The wing span is 72.2 feet with a sweep angle of 25° at the 25% chord and a wing area of 725 square feet. The wing is a full cantilever construction with a centersection mounted on the upper portion of a Gulfstream II type airframe. The wing consists of:

a. Main box structure

b. Leading edge structure (fixed)

c. Leading edge high-lift flaps (Krueger)

d. Trailing edge structure (fixed)

e. Trailing edge high-lift flaps (Tripple slotted)

f. Spoiler system

g. Aileron system

The main (structural) wing box consists of two spars and an upper and lower stringer reinforced skin. The front spar is located at 15% and the rear spar at 45% of the wing chord. Primary ribs are provided at structural load points as well as intermediate ribs for contour control and skin panel stabilization. Structural load point locations are the trailing edge flap tracks and actuators, aileron hinges and actuators, leading edge flap hinges and actuators, and engine pylons.

The wing box is a "state-of-the-art" fabricated assembly with provisions for attachment of leading and trailing edge fixed structure, leading and trailing edge high-lift devices, spoilers, and aileron systems. Attachment of wing to fuselage is provided by two machined fuselage bulkheads: one forward and one aft of the structural wing box. The wing general arrangement is depicted by drawing PD-111-2-003 and the structural arrangement is depicted by drawing PD-111-2-006.

High-Lift Devices

Leading Edge Krueger Flaps

Leading edge Krueger flaps were designed to sufficient detail in order to determine their dynamic load inputs to the wing box. These inputs will be used as parameters in a computer flutter analysis program.

The flap chords were described as being 25% of the wing chord outboard of the engine pylons and 15% of the wing chord inboard and between the engine pylons. Extended position is to be 60° to the wing reference plane. With these inputs, a leading edge section was

drawn and the mechanics of an articulated 25% chord flap was designed to retract into the wing forward of the front spar (15% chord). In addition, a one piece 15% chord flap was drawn.

Following a review by NASA, this concept was detailed on drawings PD-111-2-004 and PD-111-2-005. Actuators and hinge points, as well as spanwise trim lines, were determined utilizing the previously established wing box ribs as hard points where possible. This information is depicted on planform general arrangement drawing PD-111-2-003.

Trailing Edge Flaps

The trailing edge flap system consists of an inboard flap, center flap, and outboard flap. Each flap is made up of three (3) elements and is a triple-slot modified Fowler type. The first and second elements are the St. Cyr aerodynamic profile and the third element is a NACA 4412 profile modified to match the supercritical wing trailing edge. The chords of the three (3) elements are 10%C, 20%C and 22.5%C, respectively. The trailing edge flap structure is depicted on drawings PD-111-007 and PD-111-2-011.

Data supplied by NASA dictated flap element deflections for a landing position setting and a take-off position setting with the three (3) slots held constant at .015C (see drawing No. PD-111-2-010, flap track station 84.90 take-off position,and flap track station 84.90 landing position). The third element rotates $\pm$ 20° as flapron when positioned to the landing setting. The .015C slot remains constant throughout the $\pm$ 20° flapron rotation.

The flaps translate between retracted and take-off positions with a conventional Fowler type motion. However, between landing position and take-off position, all flap elements rotate about a common fixed point (see drawing No. PD-111-2-010, notes 1 through 4).

Engine Pylon

Support of the TF-34 engines is provided by engine pylons cantilevered forward of the structural wing box and attached to the front spar. The pylon consists of:

a. Forward engine/pylon mount fitting

b. Pylon box structure

c. Rear engine/pylon mount fitting

d. Upper and lower splice fittings (pylon to wing box)

Drawing PD-111-2-008 depicts the structural arrangement of the closed box pylon.

The forward engine/pylon mount consists of a machined forged fitting attached to the box structure. A thermal expansion link is provided between this fitting and a spherical bearing support on the engine. The rear engine/pylon mount is also a machined forged fitting. An

integral machined pin engages the engine mount for thrust and side loads and two stabilizing links resist rotation about the longitudinal axis.

Structural attachment to the wing box is provided by two upper and two lower splice fittings. Upper splice fittings are bathtub type with tension bolts. Lower splice fittings transfer load through shear into the lower wing skin.

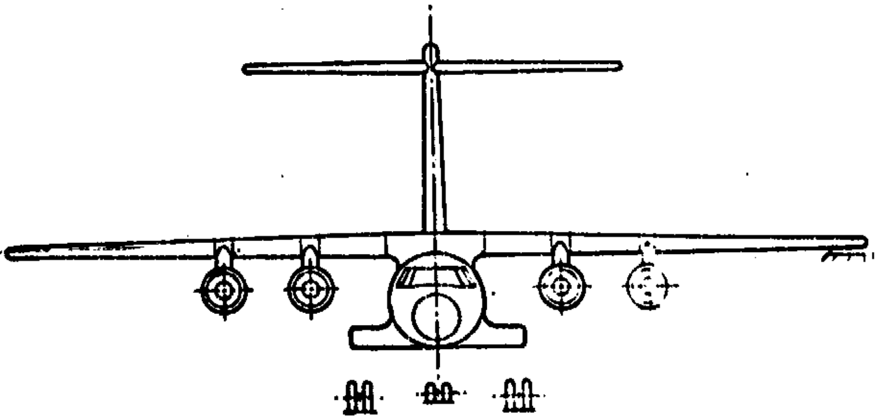
General Arrangement

A three view of the NASA EBF model is depicted on drawing PD-111-2-021.

FOLDOUT FRAME

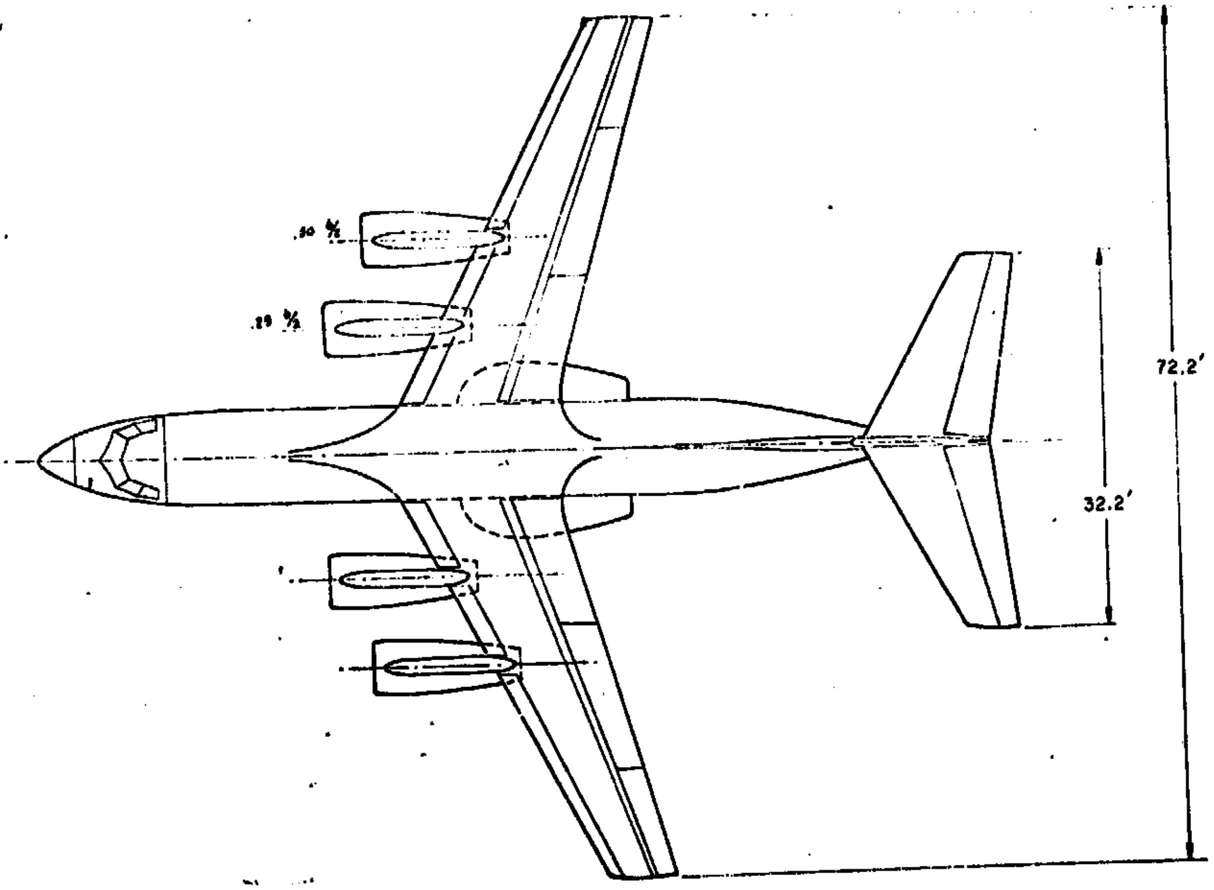
GEOMETRY	WING	HORIZ	VERT
AREA S FT^2	725.0	260.5	203
SPAN b FT	72.2	32.2	22.5
ASPECT RATIO	7.2	4.0	1.65
SWEEP $\Lambda_{1/4c}$	25°		
TAPER RATIO λ	.4	.4	.75
t/c	17% SC		
MAC FT	10.5	8.6	11.0
VOLUME COEF $\bar{V}$		1.27	.115

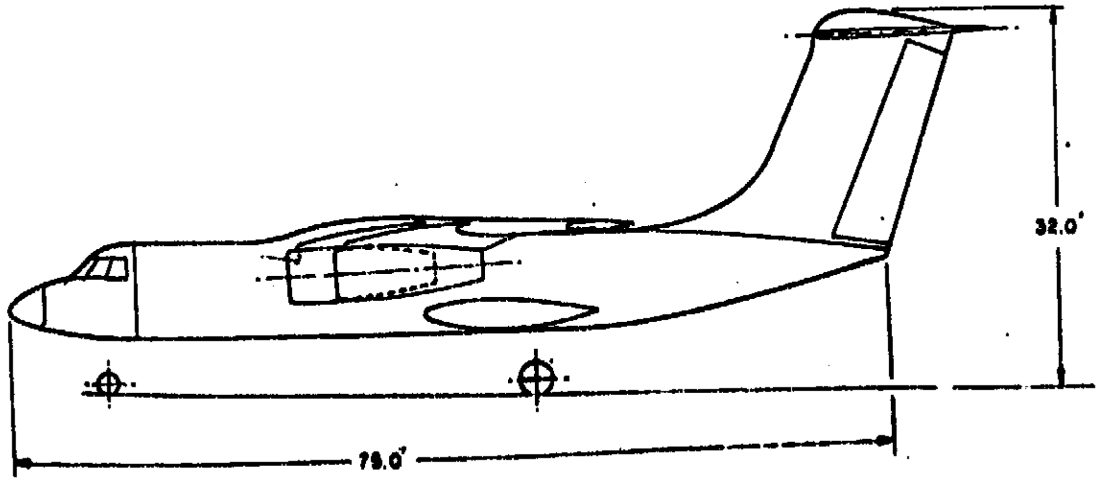
ENGINE	TF 34
RATED THRUST, LBS.	9,100
NOMINAL GROSS WEIGHT, LBS.	58,000
RANGE OF GROSS WEIGHTS, LBS.	43,500 TO 72,500
NOMINAL WING LOADING, LB/FT^2	80
RANGE OF WING LOADINGS, LB/FT^2	60 TO 100



FOLDOUT FRAME

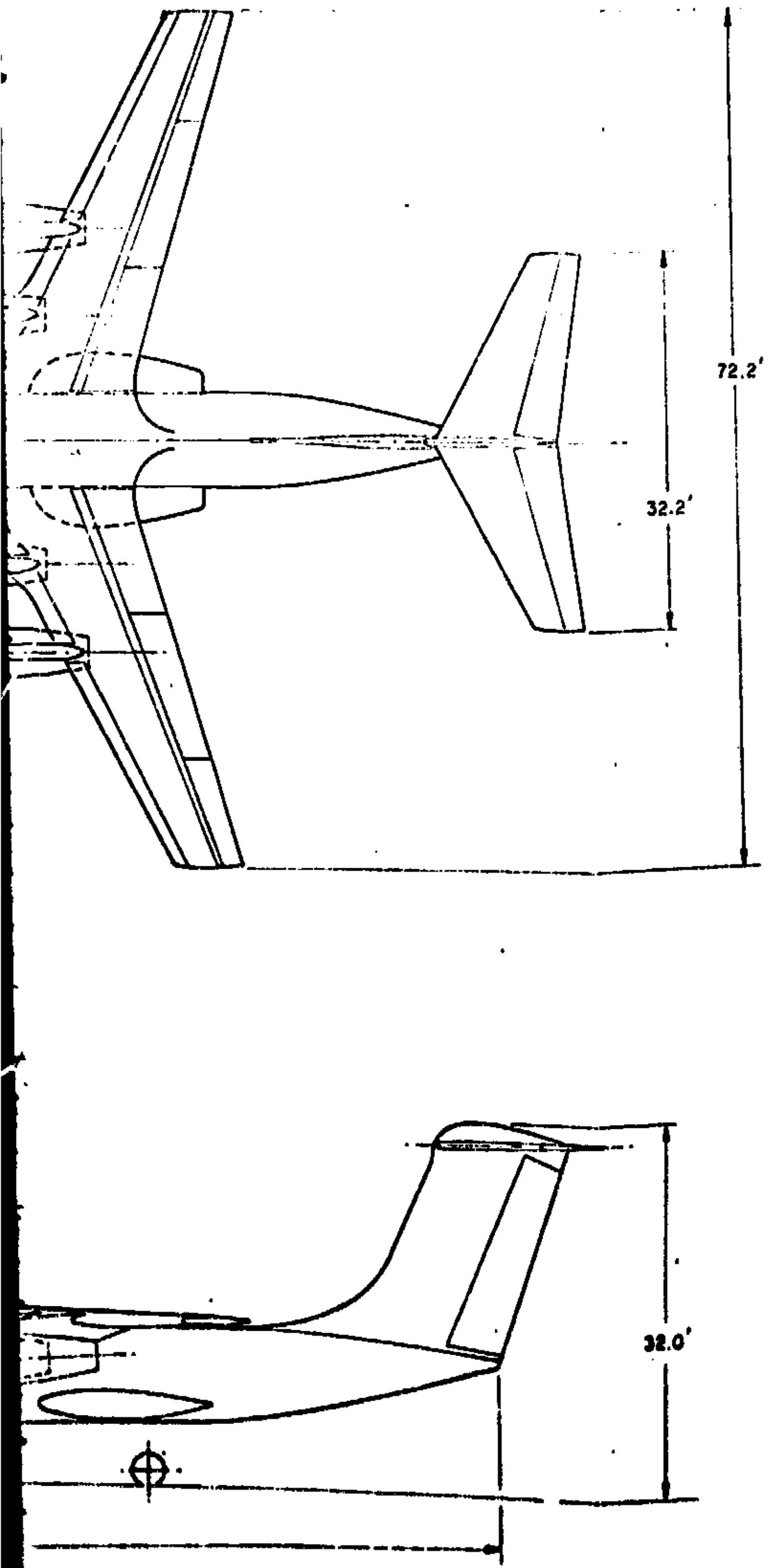
2





FOLDOUT FRAME

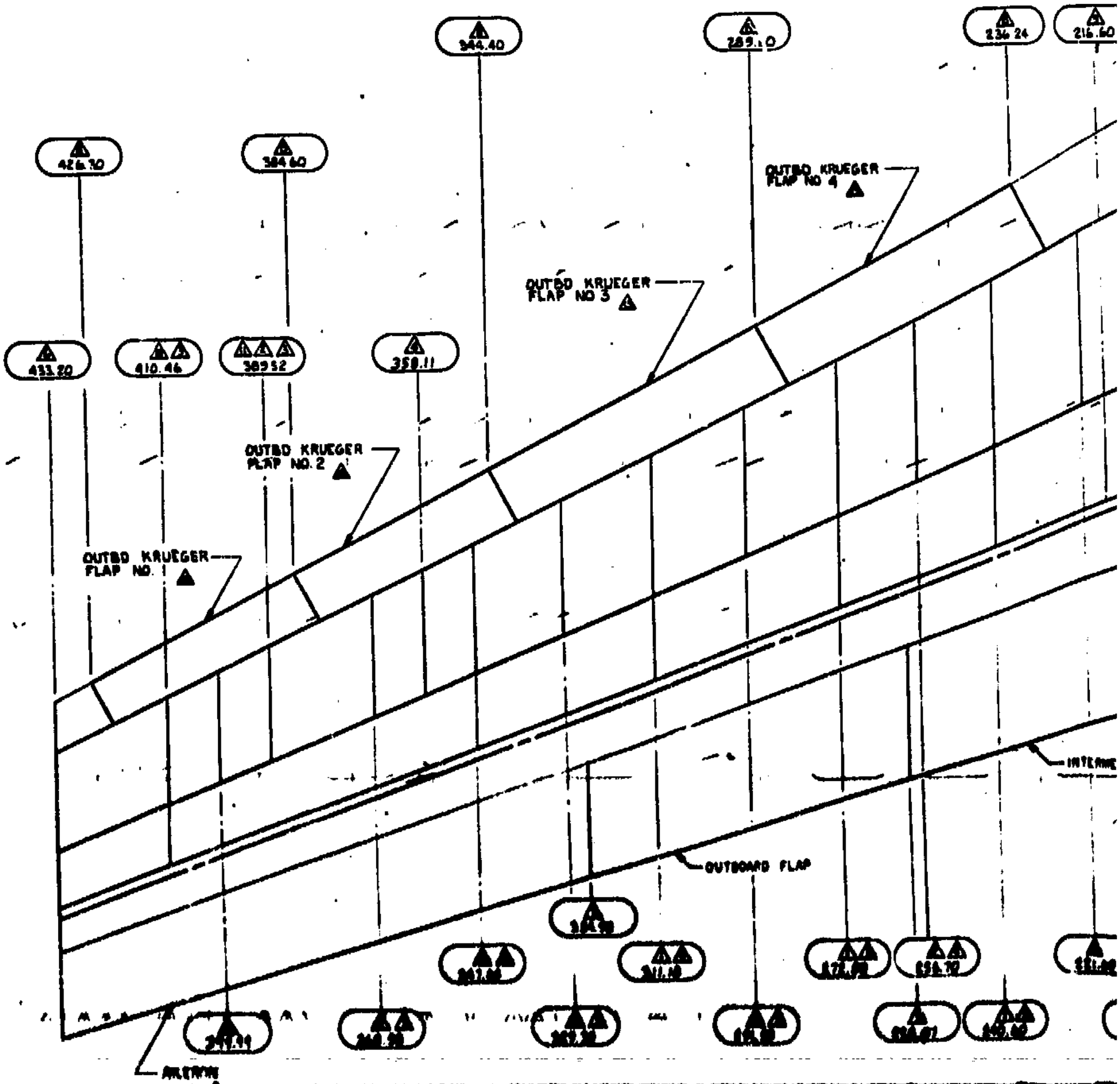
3



9

1/80	STOL
	NASA EBF MODEL
	PD-111-2-021

FOLDOUT FRAME



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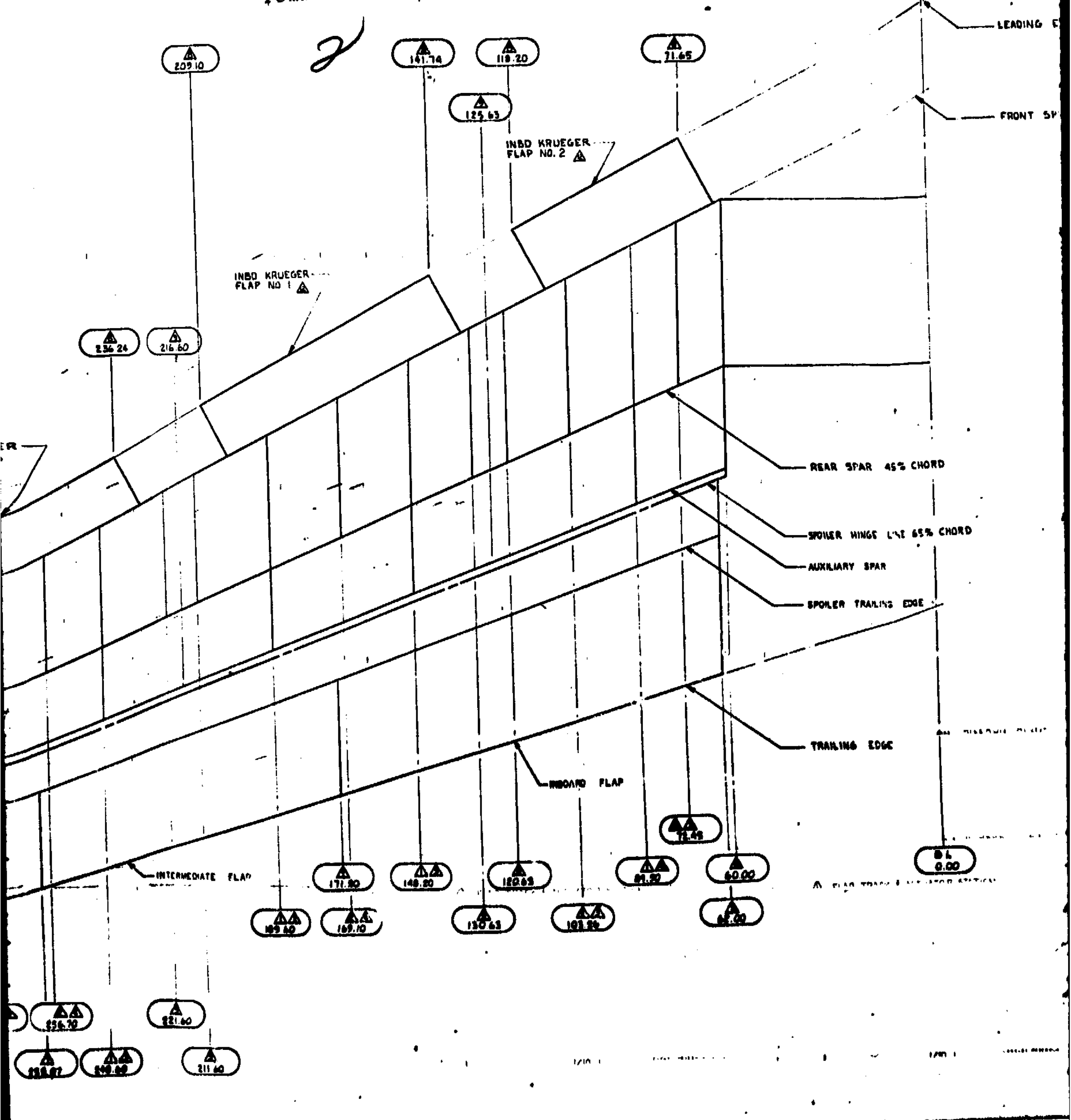

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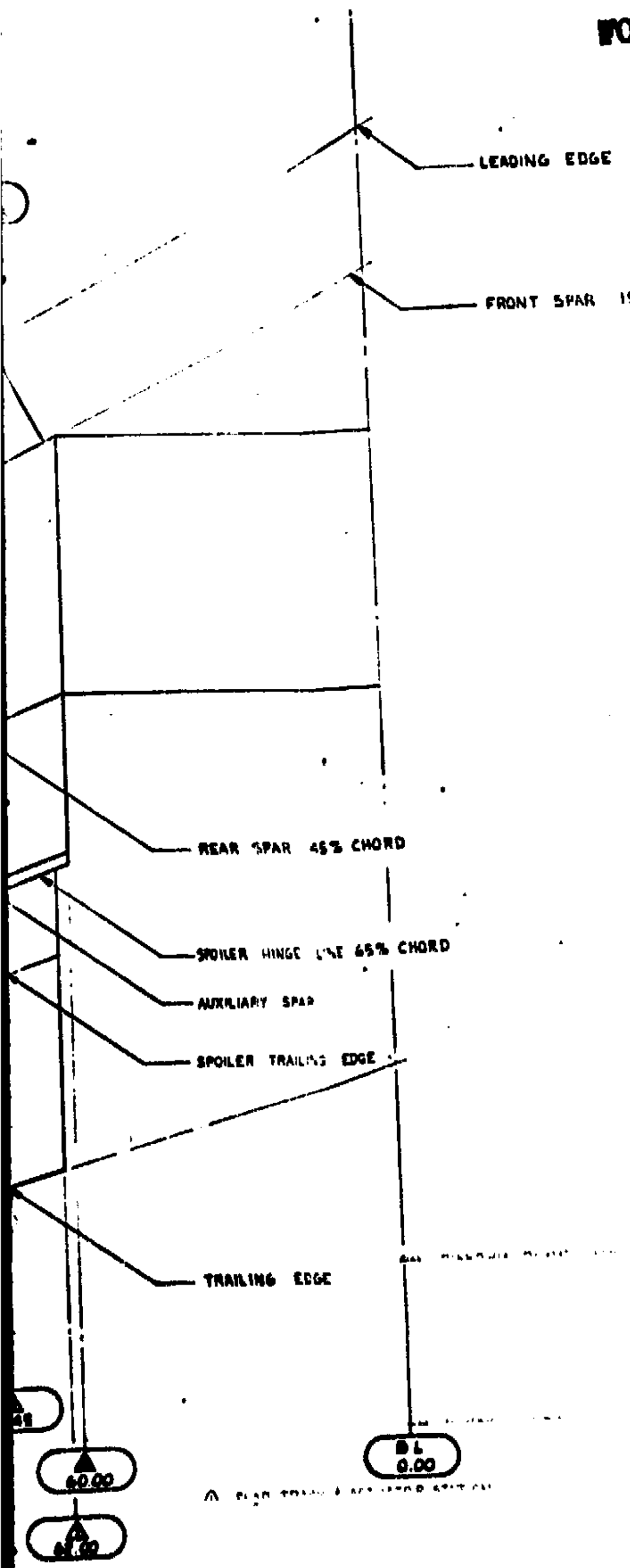
3



- OUTBOARD KRUEGER FLAP 25% CHORD-ARTICULATED
- INBOARD KRUEGER FLAP 15% CHORD
- AILERON ACTUATOR STATION
- AILERON HINGE STATION
- PYLON ℄
- END OF KRUEGER FLAP
- END OF FLAP
- CLOSEOUT RIB STATION
- PYLON RIB STATION
- KRUEGER FLAP ACTUATOR STATION
- KRUEGER FLAP HINGE STATION
- INTERMEDIATE RIB STATION
- FLAP TRACK ℄ ACTUATOR STATION

NOTES:

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STOL

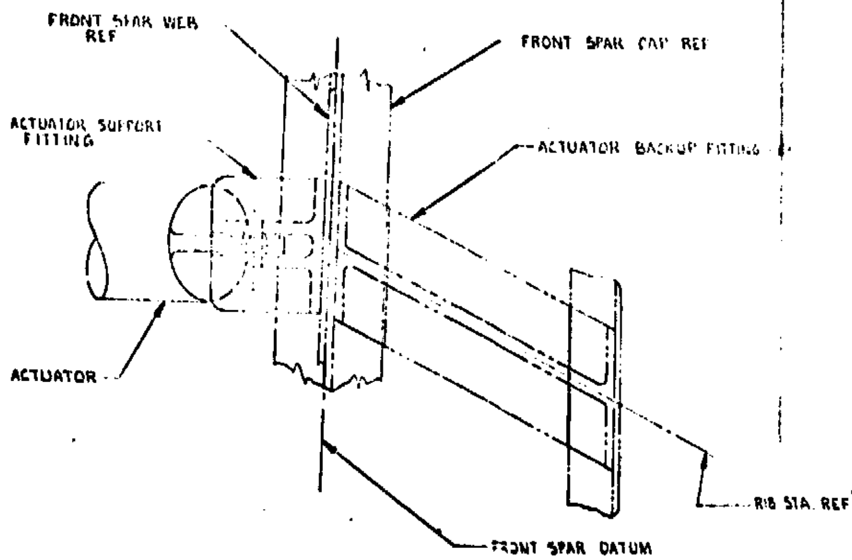
GENERAL ARRANGEMENT WING

PD-111-2-003

PD-111-S-003

FOLDOUT FRAME

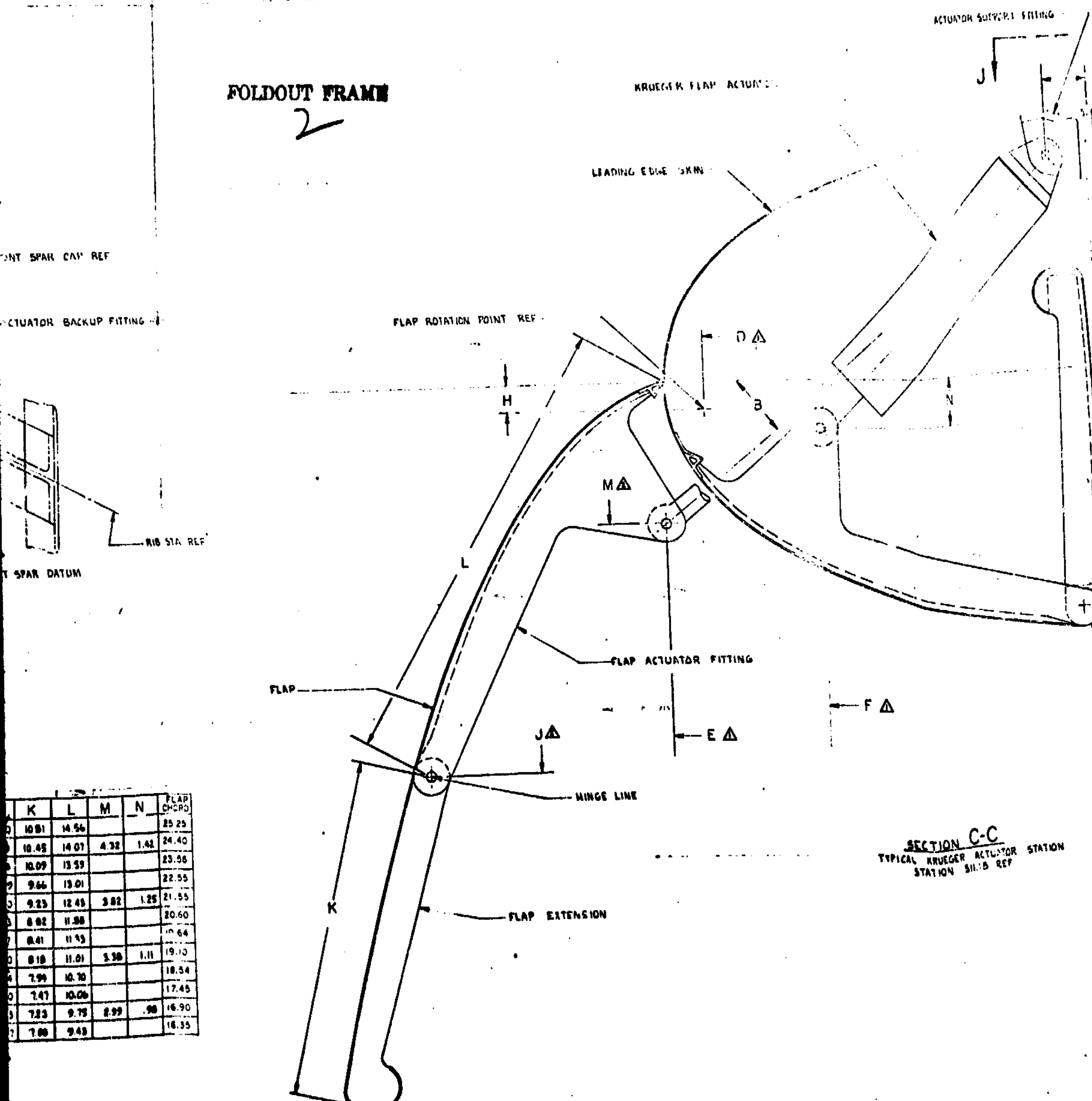
1



SECTION J-J

	A	B	WING CHORD	D	E	F	G	H	[illegible]	K	L	M	N	FLAP CHORD
HINGE STA 240.60			114.07	13.86				.90	2.50	10.81	14.56			25.25
ACTUATOR STA 256.70	1.25	1.9	110.26	13.42	14.72	9.76	6.47	.87	.89	10.45	14.07	4.32	1.42	24.40
HINGE STA 272.80			106.45	12.96				.84	.46	10.09	13.59			23.56
HINGE STA 291.99			101.91	12.40				.81	3.99	9.66	13.01			22.55
ACTUATOR STA 311.18	1.25	1.66	97.37	11.85	13.0	8.62	5.71	.77	3.50	9.23	12.43	3.82	1.25	21.55
HINGE STA 329.39			93.06	11.33				.74	3.03	8.82	11.88			20.60
HINGE STA 347.64			88.74	10.80				.70	2.57	8.41	11.33			[illegible] 64
ACTUATOR STA 358.11	1.25	1.49	86.26	10.50	11.52	7.64	5.05	.68	2.30	8.18	11.01	3.38	1.11	19.10
HINGE STA [illegible]			83.79	10.20				.66	2.04	7.94	10.70			18.54
HINGE STA 389.98			78.84	9.60				.62	1.50	7.47	10.06			17.45
ACTUATOR STA 399.99	1.25	1.32	76.36	9.29	10.20	6.76	4.41	.60	1.23	7.23	9.75	2.99	.98	16.90
HINGE STA 410.90	[illegible]		73.88	8.99				.58	.97	7.00	9.43			16.35

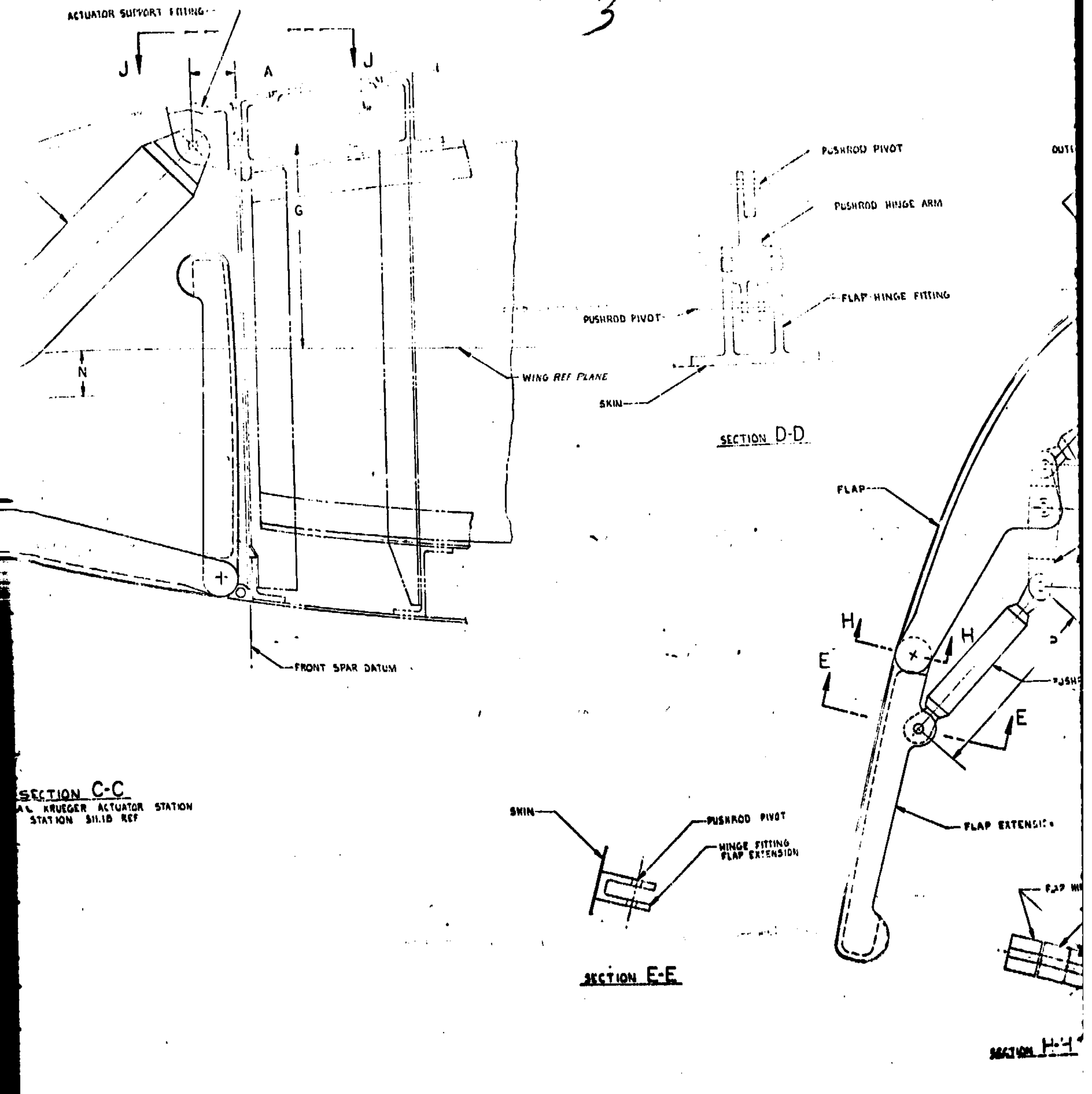
FOLDOUT FRAME 2

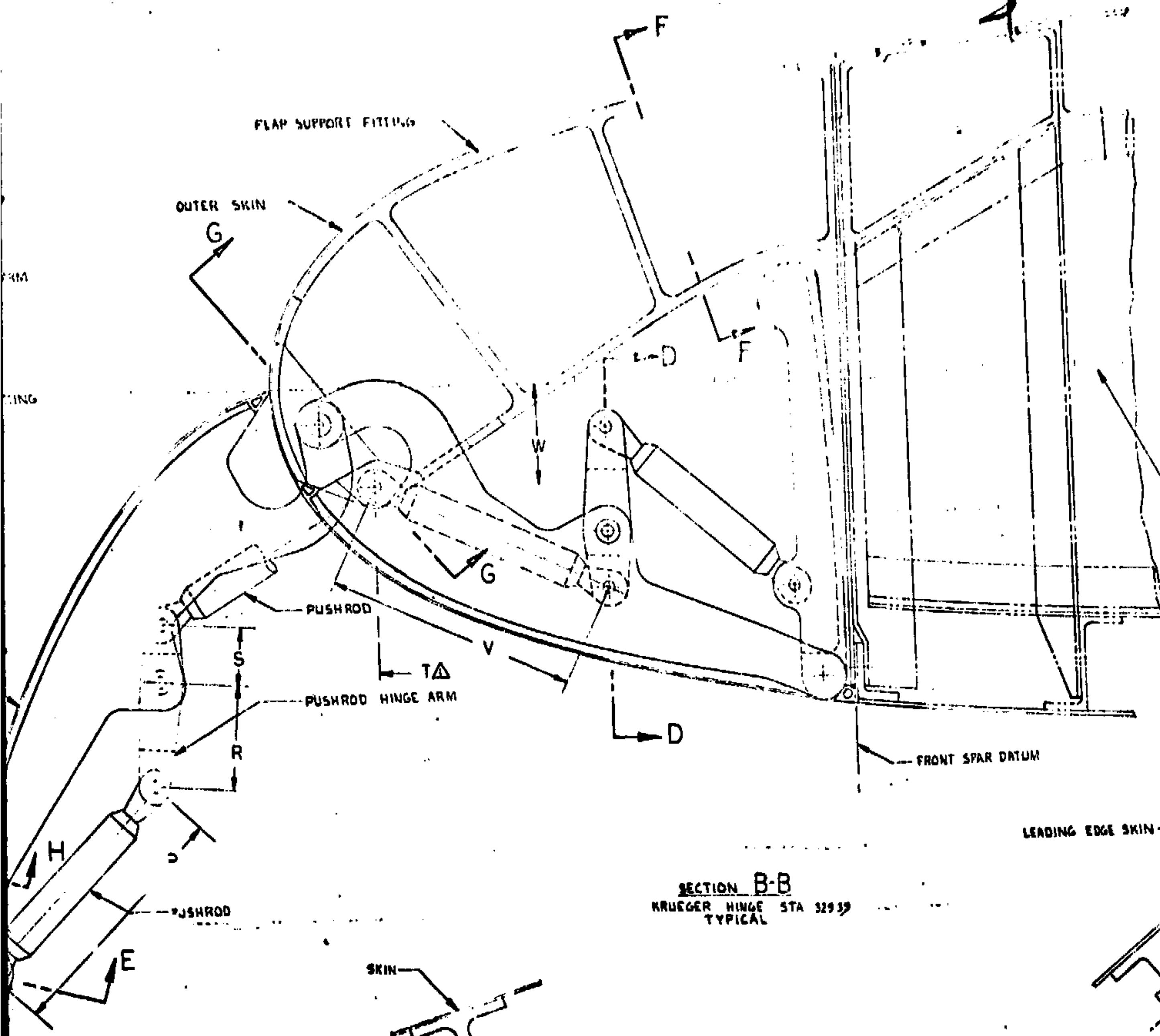


K	L	M	N	FLAP CHORD
10.81	14.56			25.25
10.45	14.07	4.32	1.42	24.40
10.09	13.59			23.56
9.66	13.01			22.55
9.23	12.45	3.82	1.25	21.55
8.82	11.88			20.60
8.41	11.53			[illegible]64
8.18	11.01	3.38	1.11	19.10
7.99	10.70			18.54
7.47	10.06			17.45
7.23	9.75	2.99	.98	16.90
7.00	9.43			16.35

SECTION C-C
TYPICAL KRUEGER ACTUATOR STATION
STATION [illegible] REF

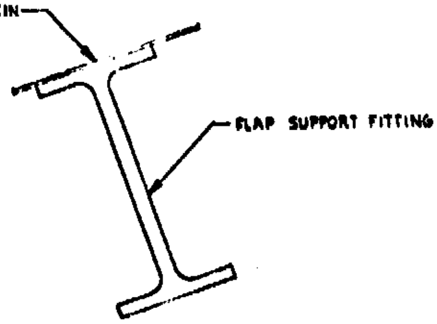
FOLDOUT FRAME 3



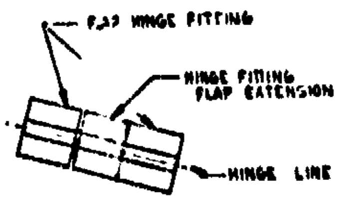


HINGE STA	WING CHORD	[illegible]	[illegible]
240.60	110.01	6.49	2.76
272.80	106.45	6.06	2.57
291.93	101.91	5.80	2.46
323.35	95.06	5.30	2.25
[illegible]	88.74	5.05	2.15
[illegible]	83.79	4.77	2.03
389.52	78.84	4.49	1.91
410.46	73.53	4.21	1.79

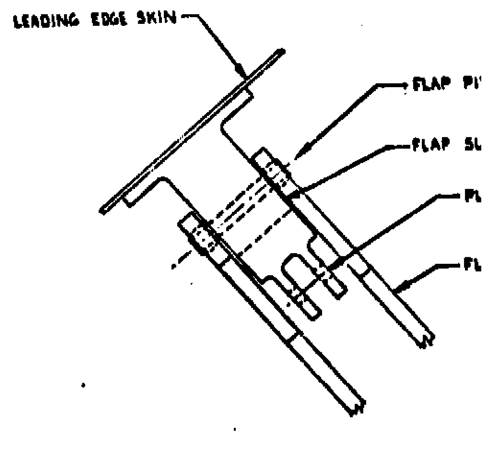
SECTION B-B
KRUEGER HINGE STA 329.39
TYPICAL



SECTION F-F
TYP FITTING SECTION



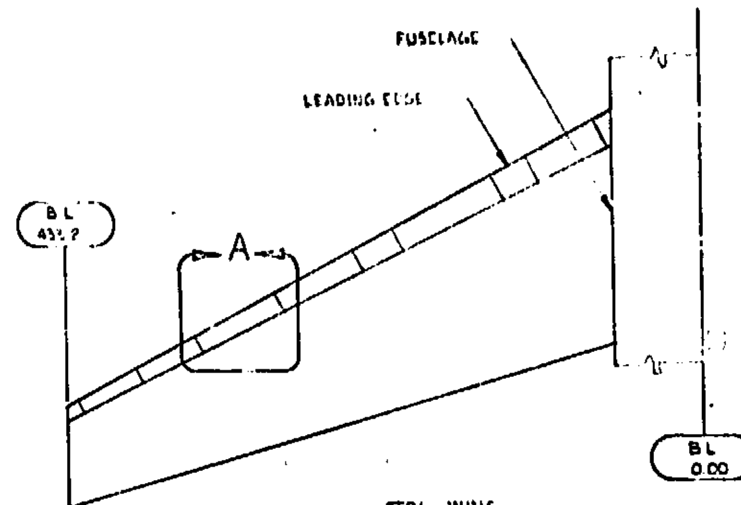
SECTION H-H



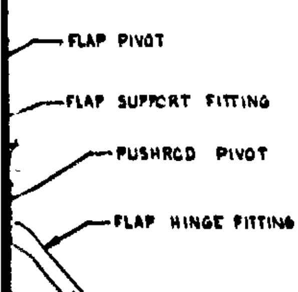
SECTION G-G

FOLDOUT FRAME 5

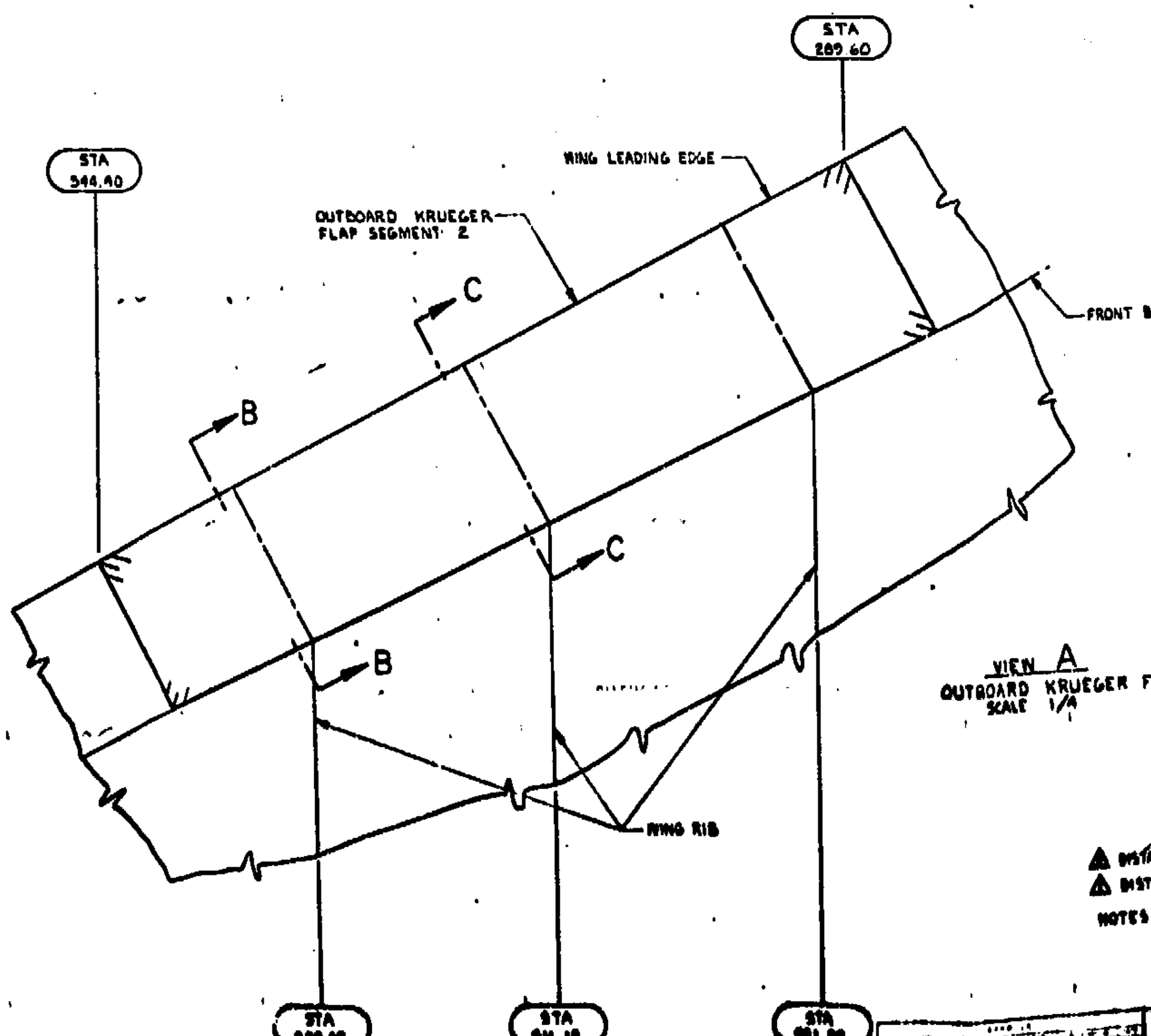
	P	R	S	T	V	W
[illegible]	6.49	[illegible]		12.42	6.56	2.57
[illegible]	[illegible]	[illegible]	1.43	11.59	6.12	2.40
[illegible]	5.80	2.46	1.37	[illegible]	[illegible]	2.30
[illegible]	5.30	2.25	1.25	10.13	5.55	2.10
[illegible]	5.05	2.15	1.19	[illegible]	5.30	2.00
79	4.77	2.03	1.13	9.12	4.82	1.87
4	[illegible]	1.91	1.06	8.59	4.53	1.78
[illegible]	4.21	1.79	.99	8.05	4.26	1.67



STOL WING
SCALE 1/40



G-G

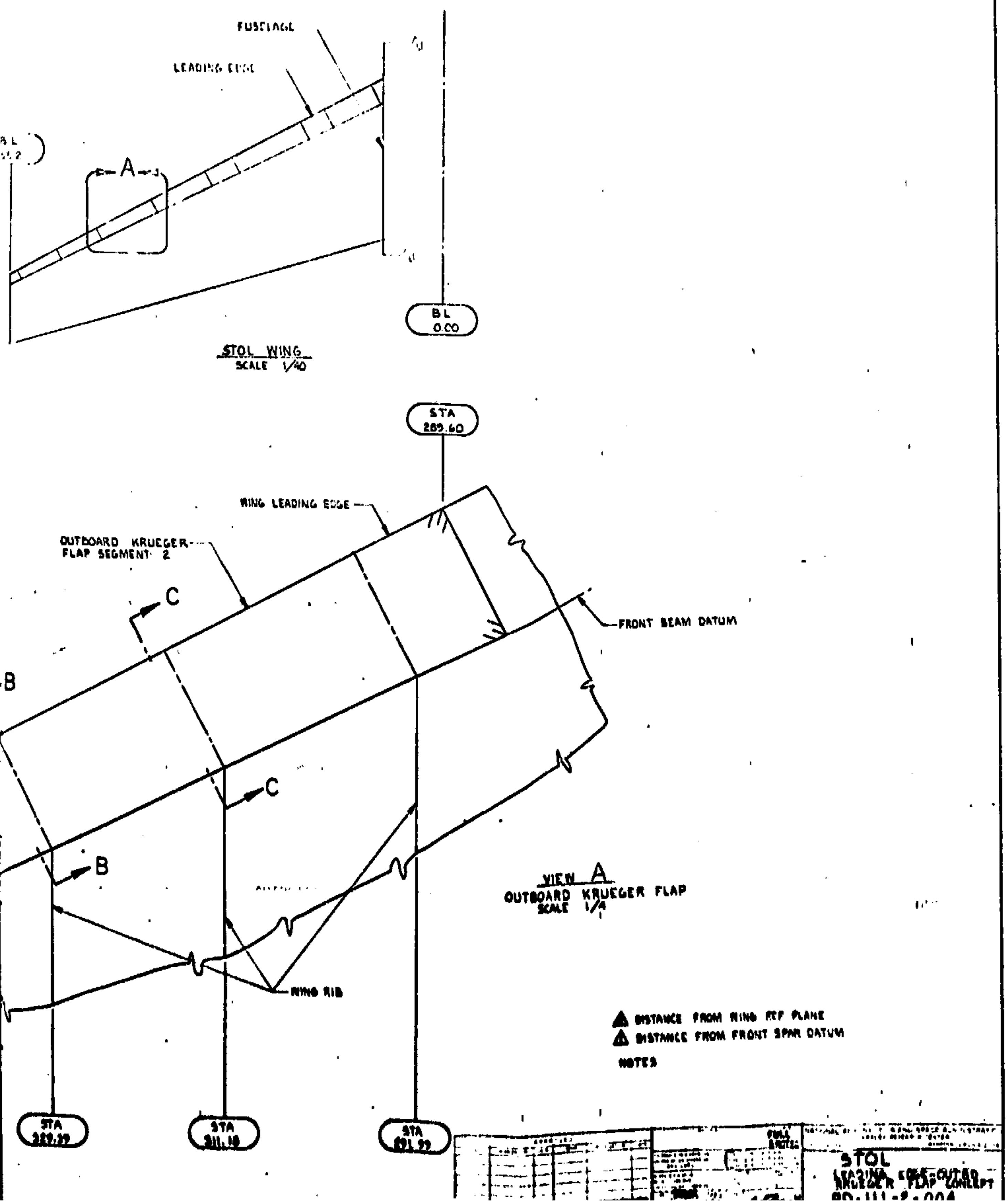


VIEW A
OUTBOARD KRUEGER FL
SCALE 1/4

NOTES

FOLDOUT FRAME 5

FOLDOUT FRAME 6



STOL WING
SCALE 1/40

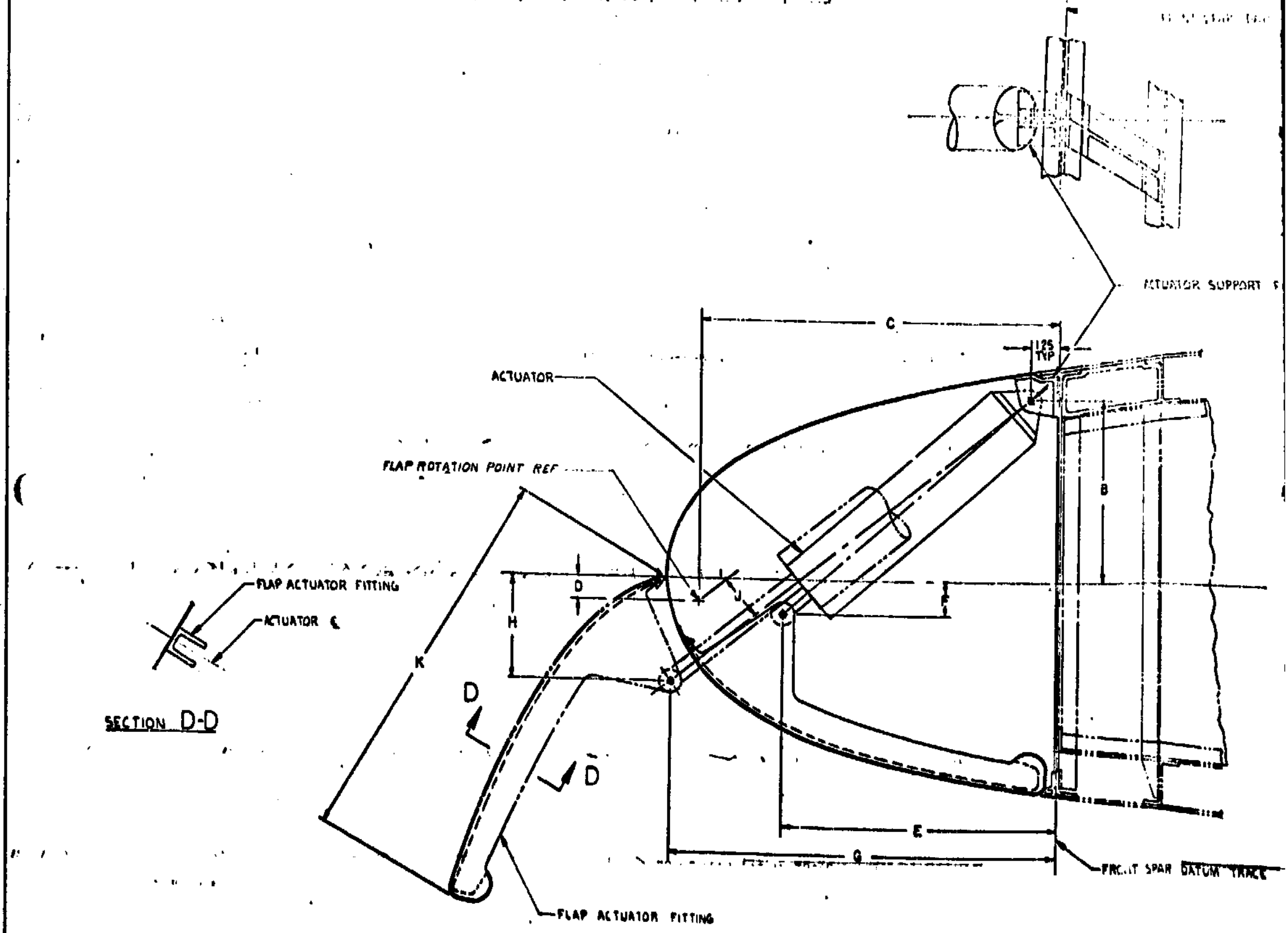
VIEW A
OUTBOARD KRUEGER FLAP
SCALE 1/4

11

111-S-004

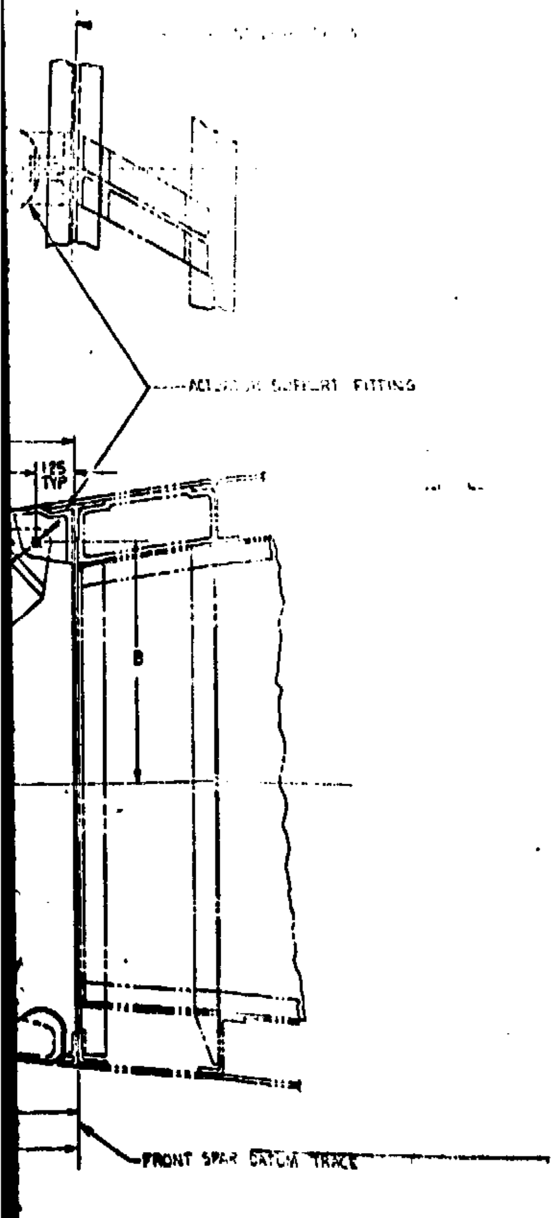
FOLDOUT FRAME

	WING CHORD	A	B	C	D	E	F	G	H	J	K	L	M	N
[illegible]	[illegible]	[illegible]		15.15	1.72	[illegible]	[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	8.42	[illegible]
[illegible]	[illegible]		[illegible]	18.47	1.10	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]			
[illegible]	[illegible]	[illegible]		17.51	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		[illegible]	2.15	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]		[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]			
[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	[illegible]	[illegible]	[illegible]			[illegible]	[illegible]	[illegible]	[illegible]

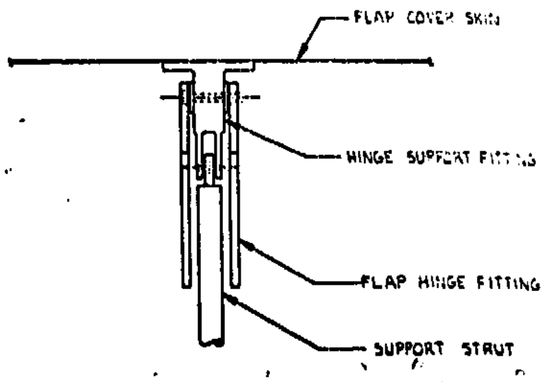


SECTION E-E
FLAP ACTUATOR STA 169.10

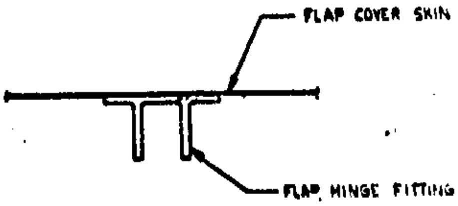
FOLDOUT FRAME 2



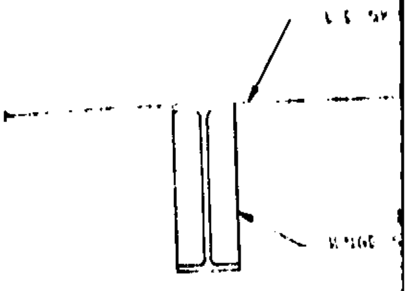
SECTION E-E
FLAP ACTUATOR STA



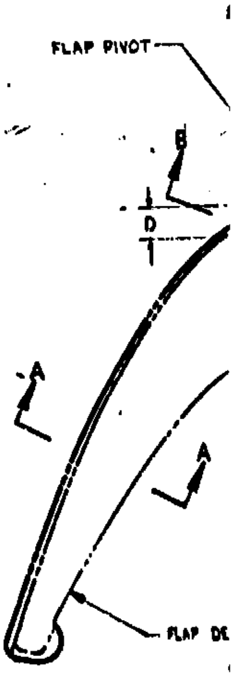
SECTION B-B



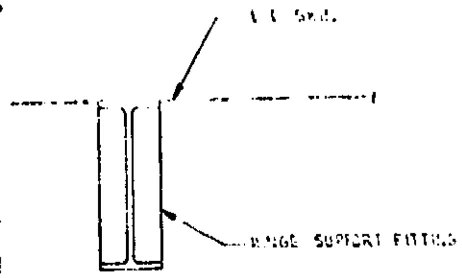
SECTION A-A



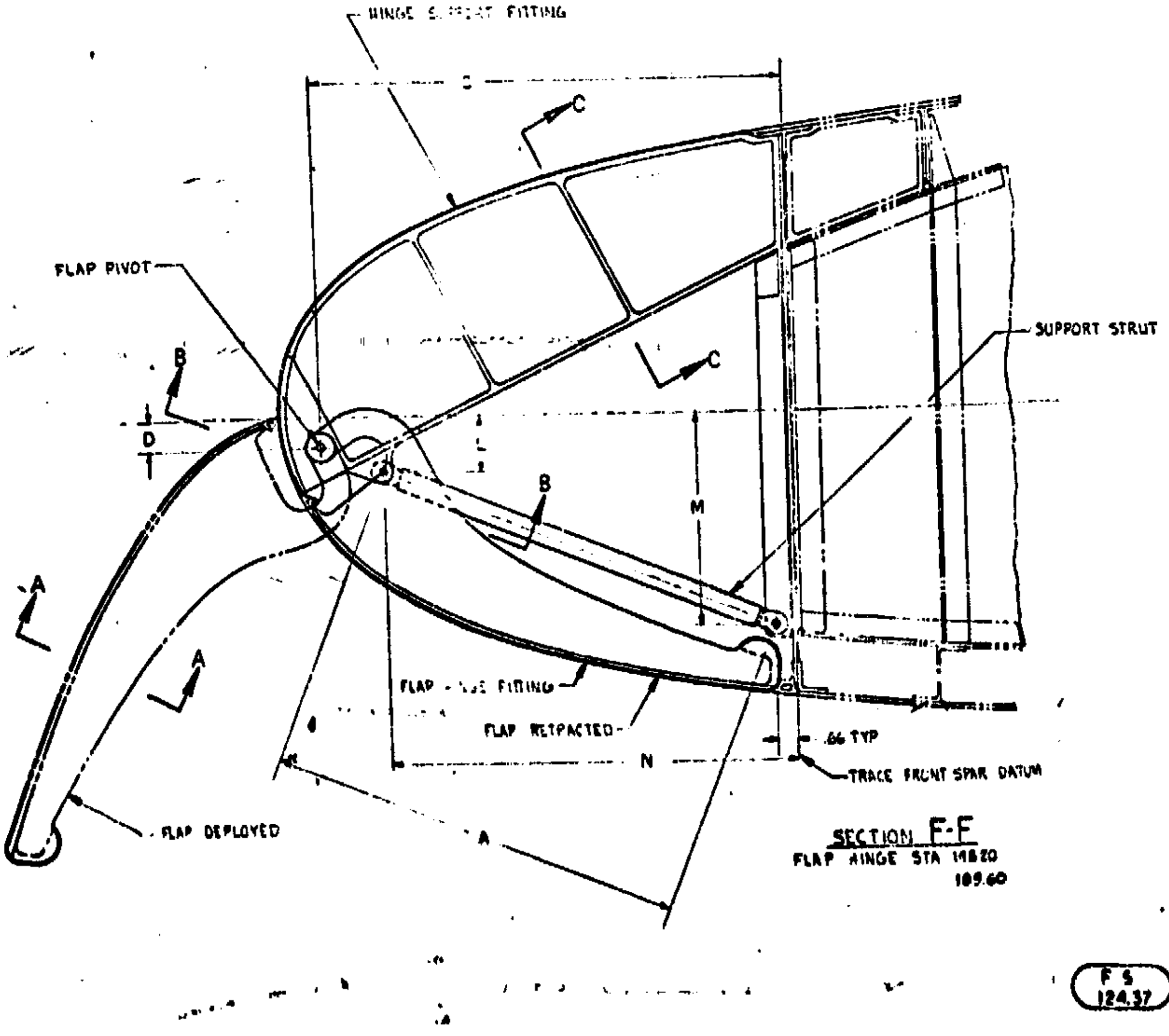
SECTION C-C



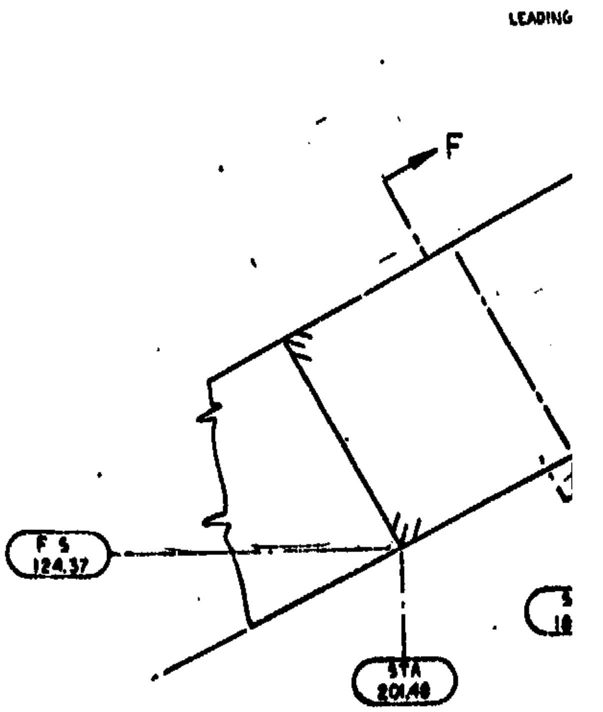
FOLDOUT FRAME
3



SECTION C-C



SECTION F-F
FLAP HINGE STA 148.20
189.60



FOLDOUT FRAME

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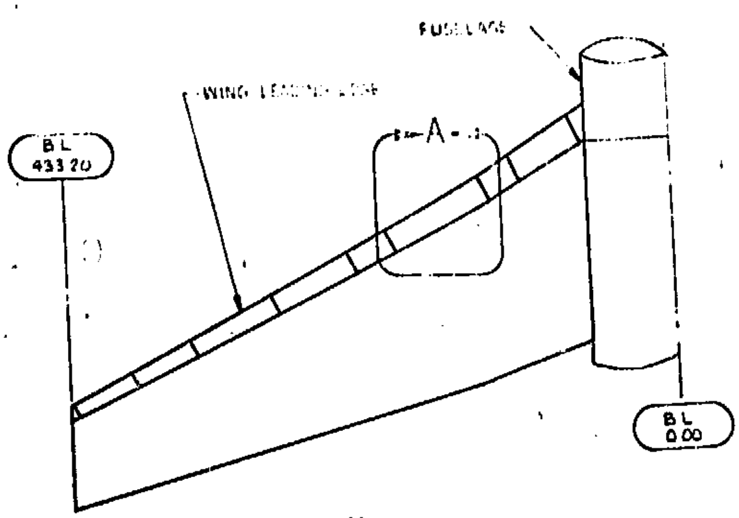
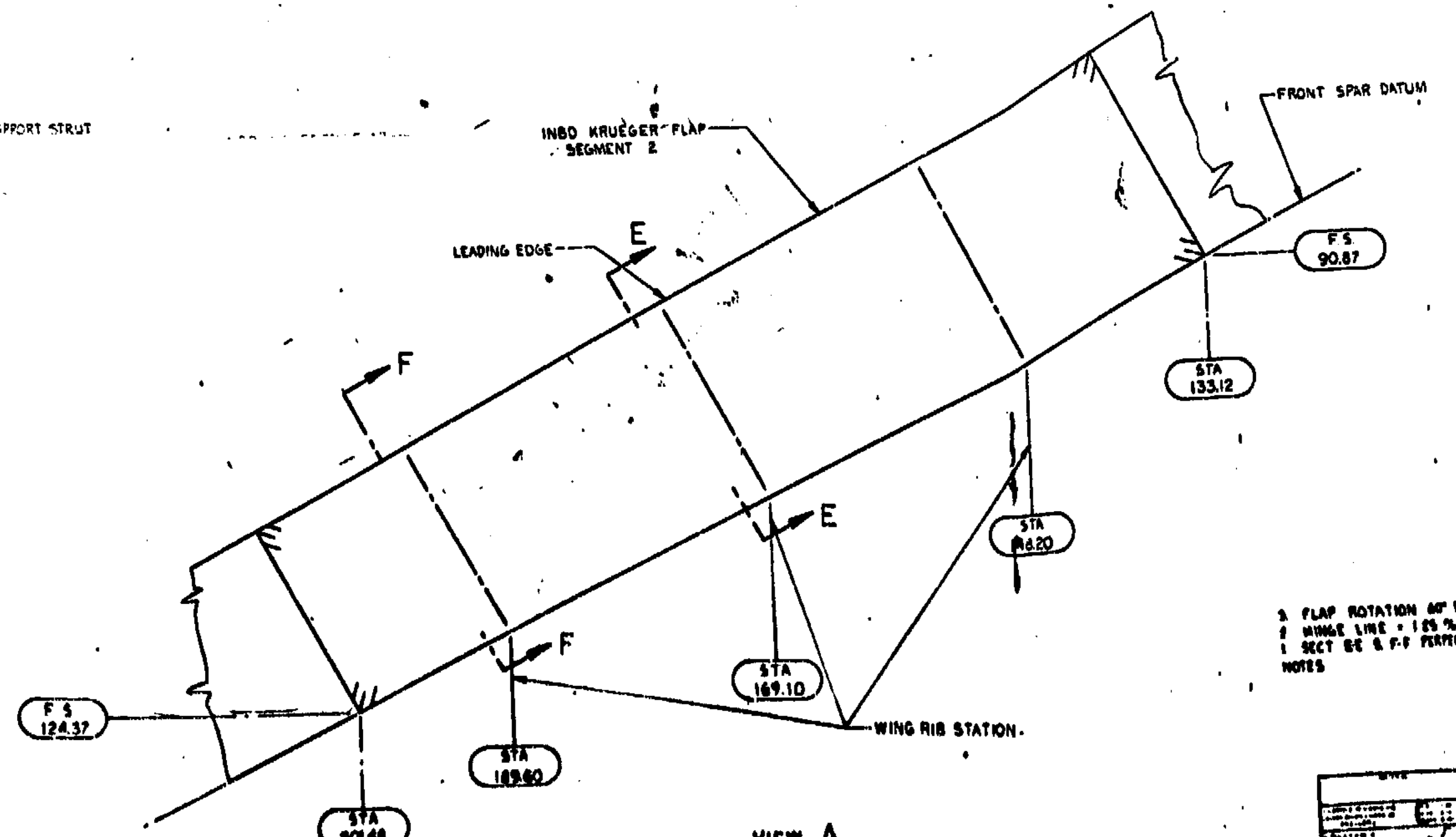
FOLDOUT



STOL WING
SCALE 1/40



VIEW A
SCALE 1/8

3. FLAP ROTATION 60° FROM REF
2. HINGE LINE = 1.25 % CHORD
1. SECT E-E & F-F PERPENDICULAR
NOTES

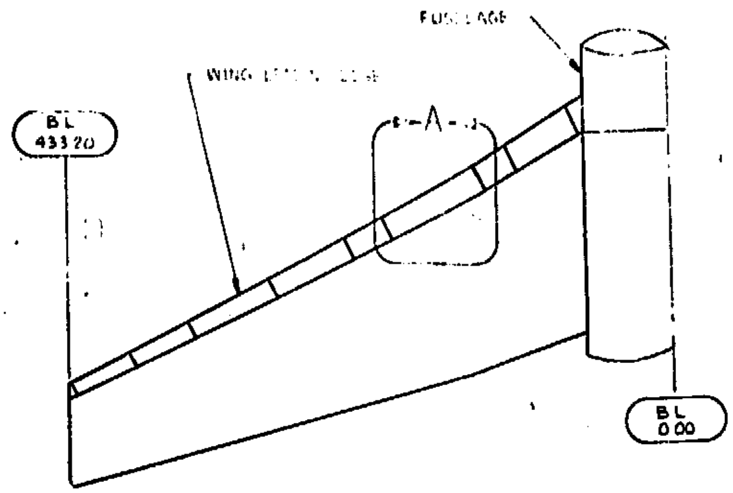
FOLDOUT FRAME

1

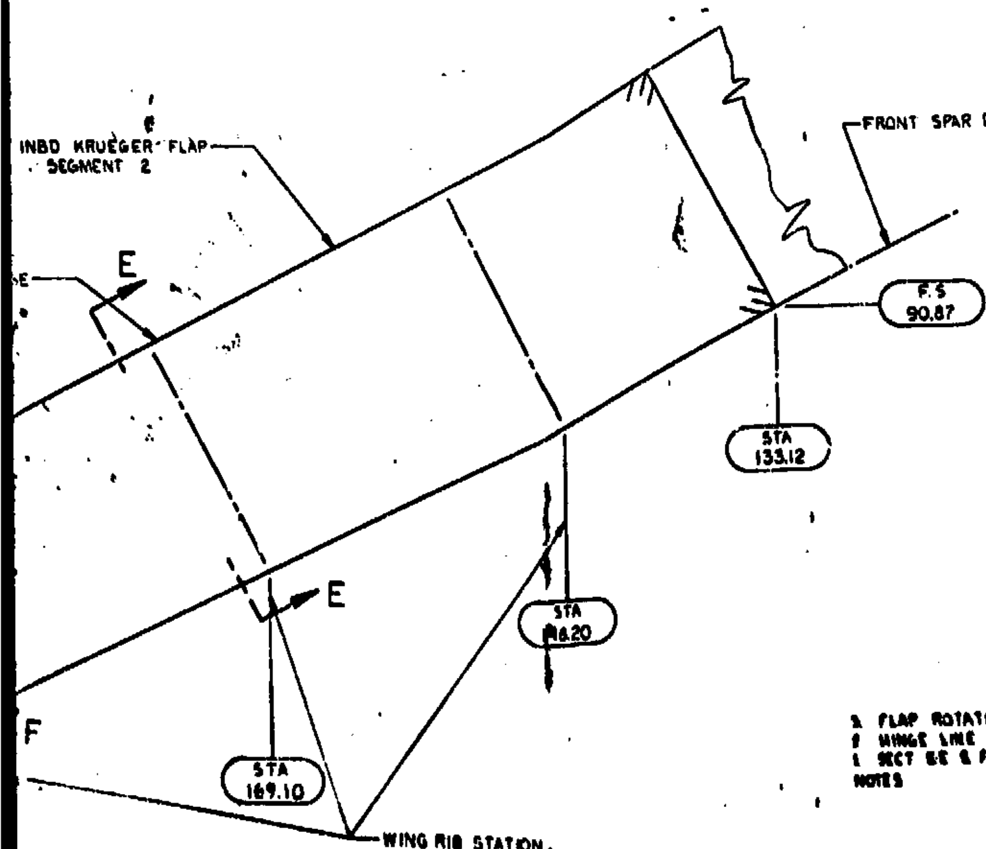
FOLDOUT FRAME

[redacted]

5



STOL WING
SCALE 1/40



3. FLAP ROTATION 60° FROM WRP EXTENDED
2. HINGE LINE - 1.25 % CHORD
1. SECT EE & F-F PERPENDICULAR TO HINGE LINE
NOTES

12

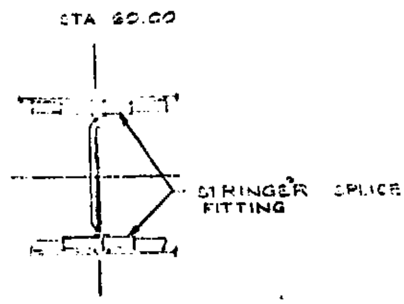
VIEW A
SCALE 1/4

STOL
LEADING EDGE KRUEGER FLAP CONCEPT
PD-111-2-005

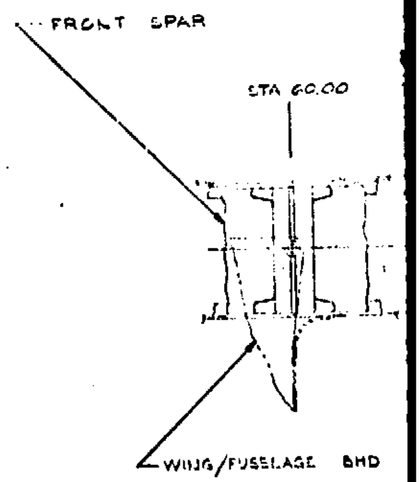
D-111-3-003

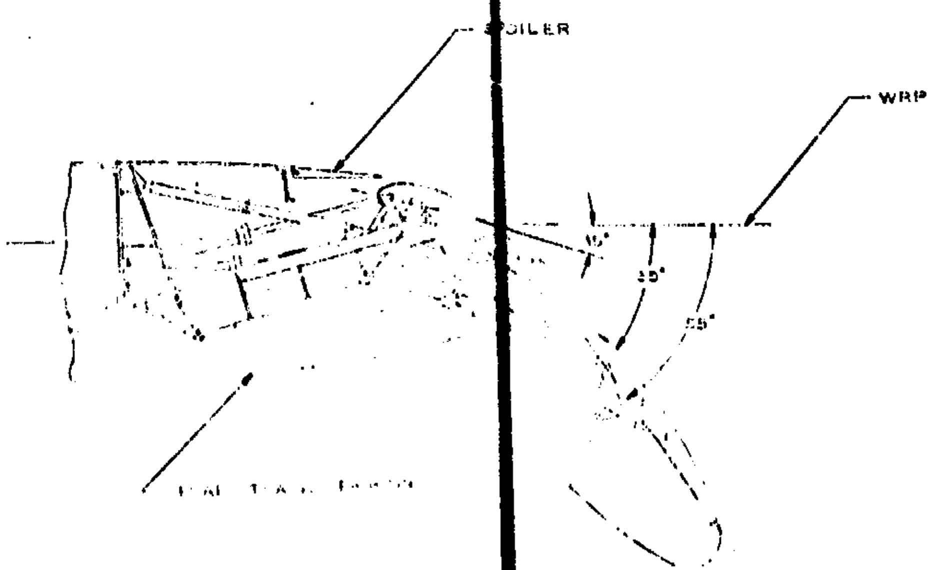
FOLDOUT FRAME

FOLDO

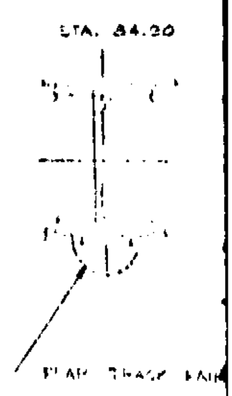


SECTION G-G





DETAIL F-F 1/10
LANDING CONFIGURATION



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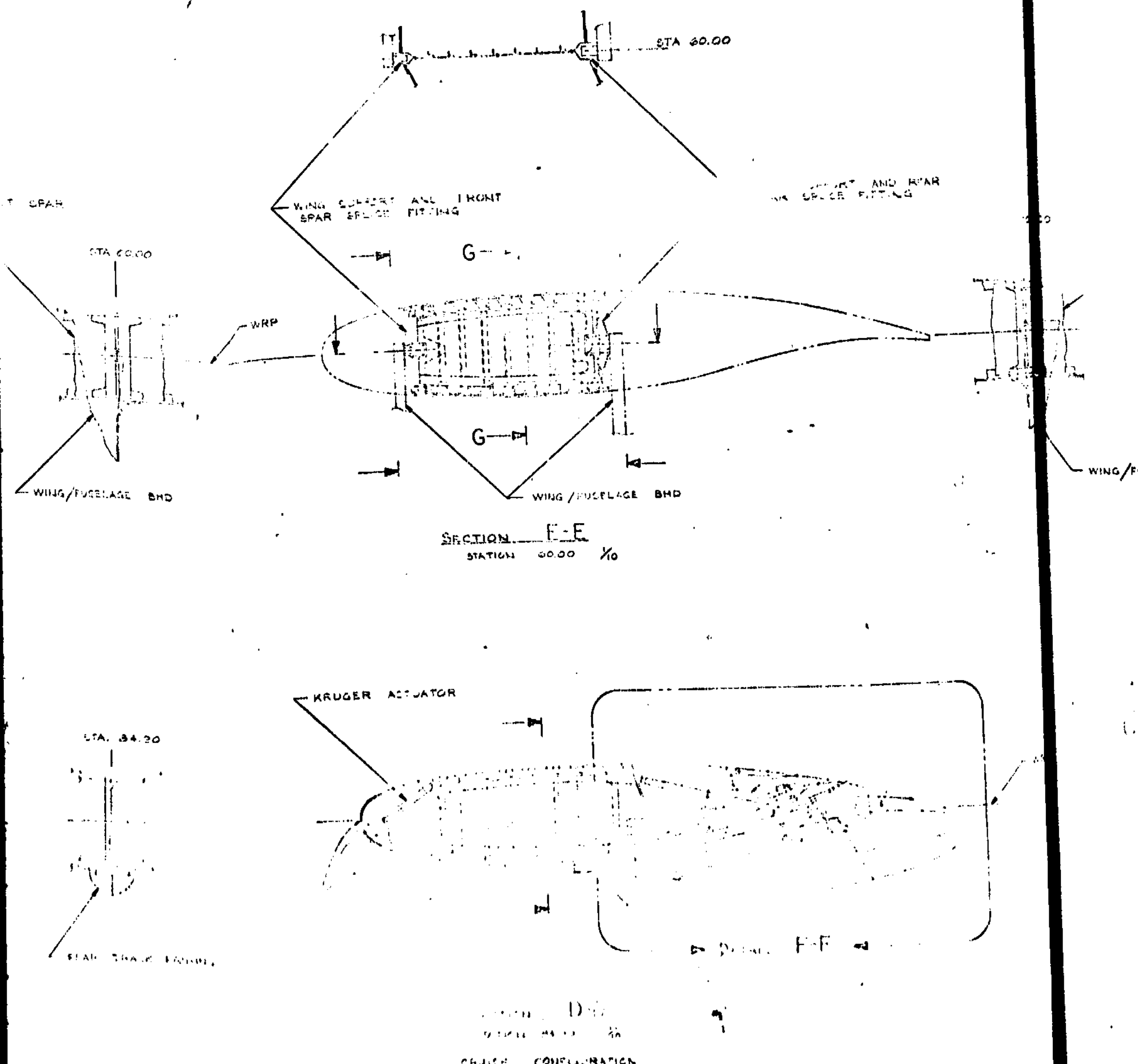

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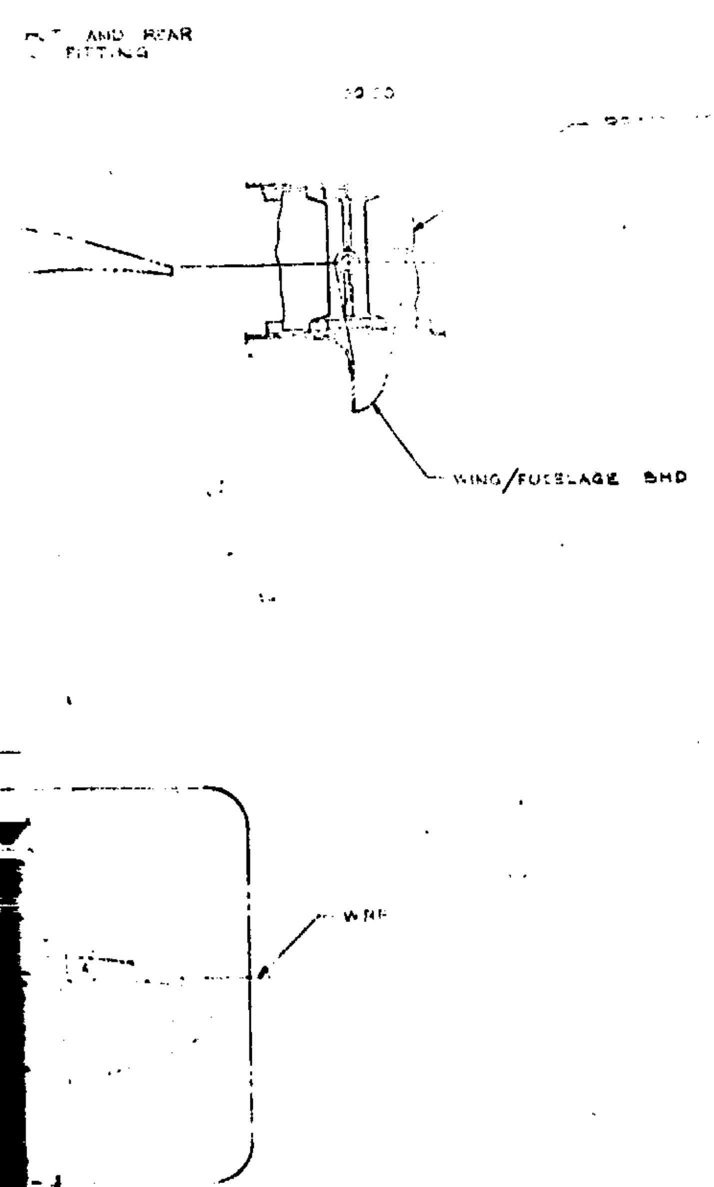
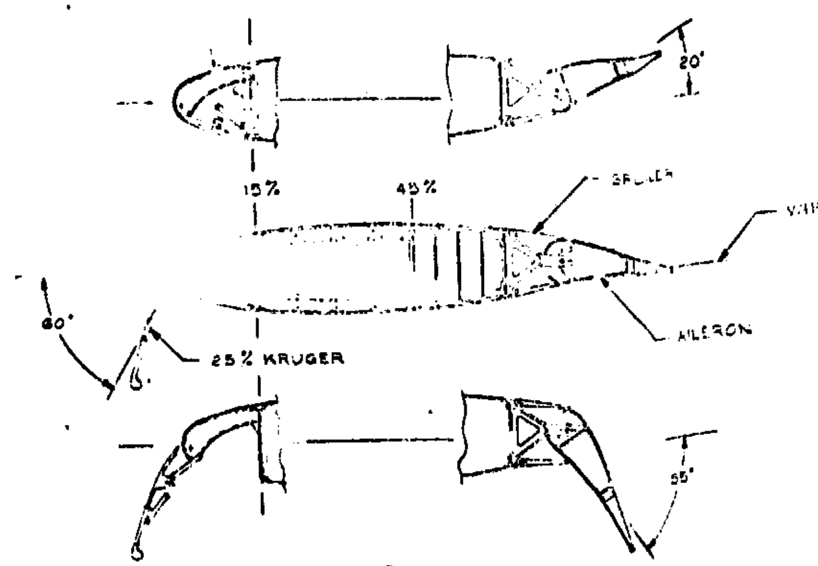
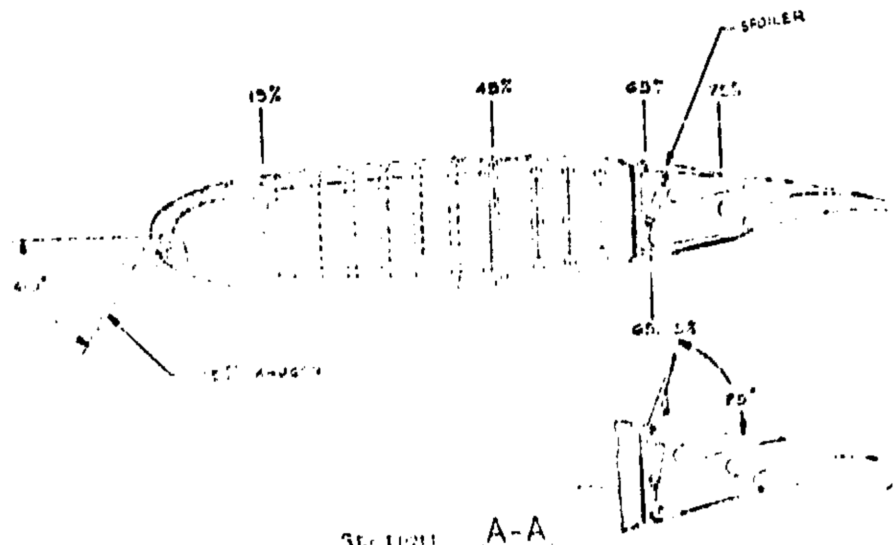


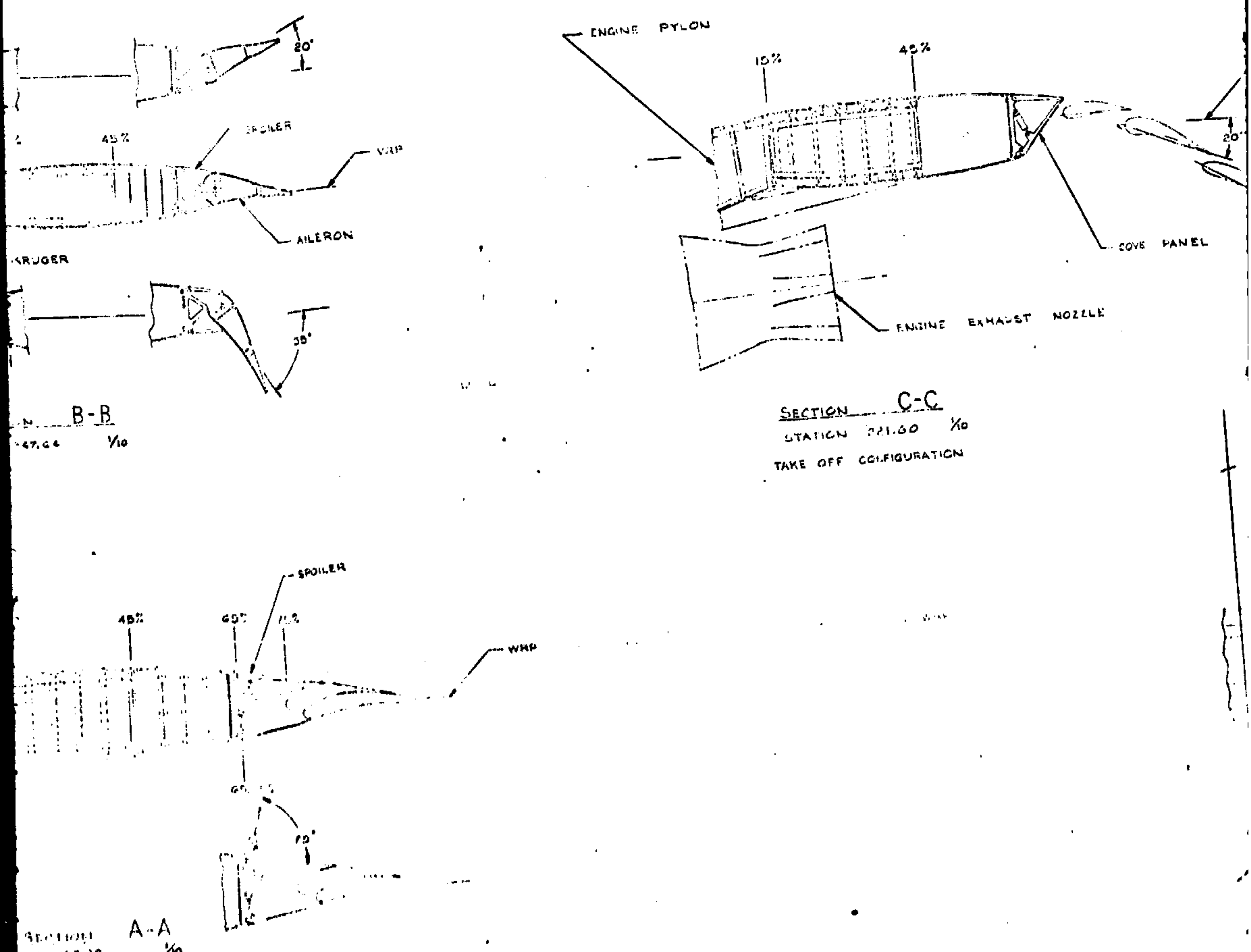


SECTION B-B
STA 547.64 1/10



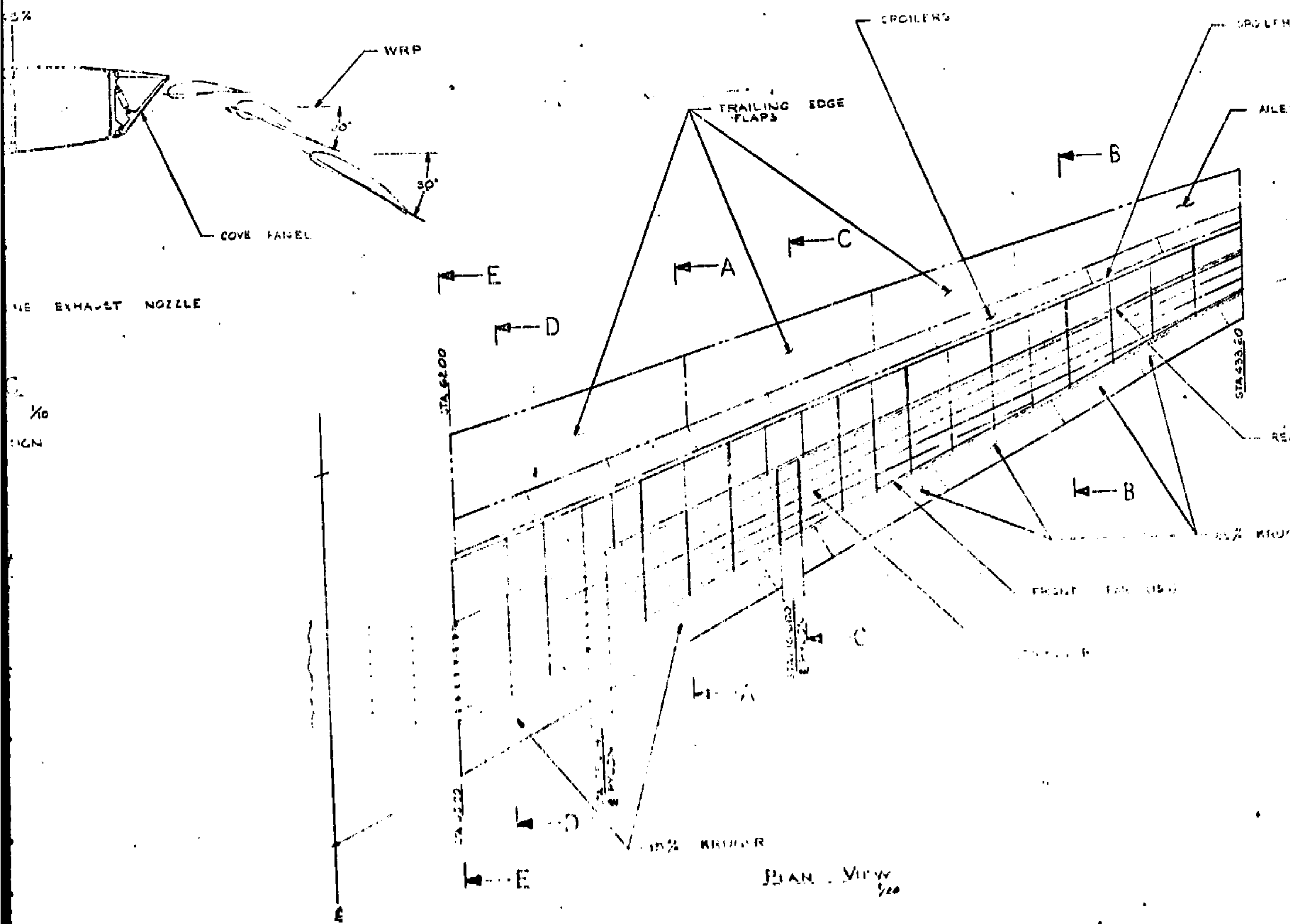
SECTION A-A
STA 169.30 1/10

FOLDOUT FRAME 4



FRAME

FOLDOUT FRAME 5



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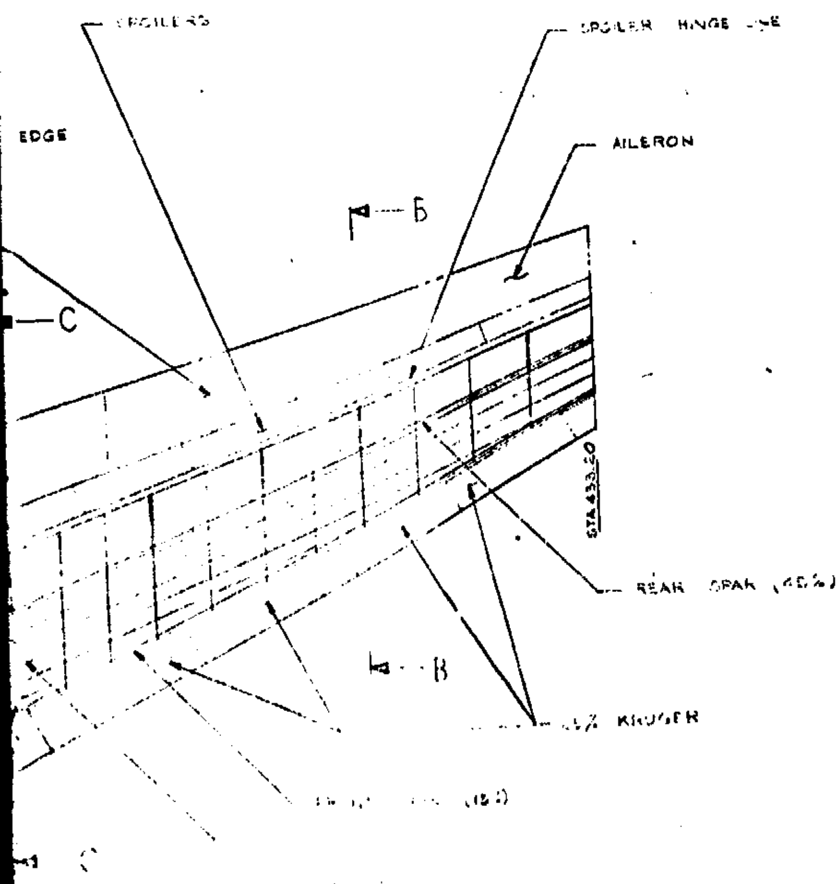
FOLDOUT FRAME 6

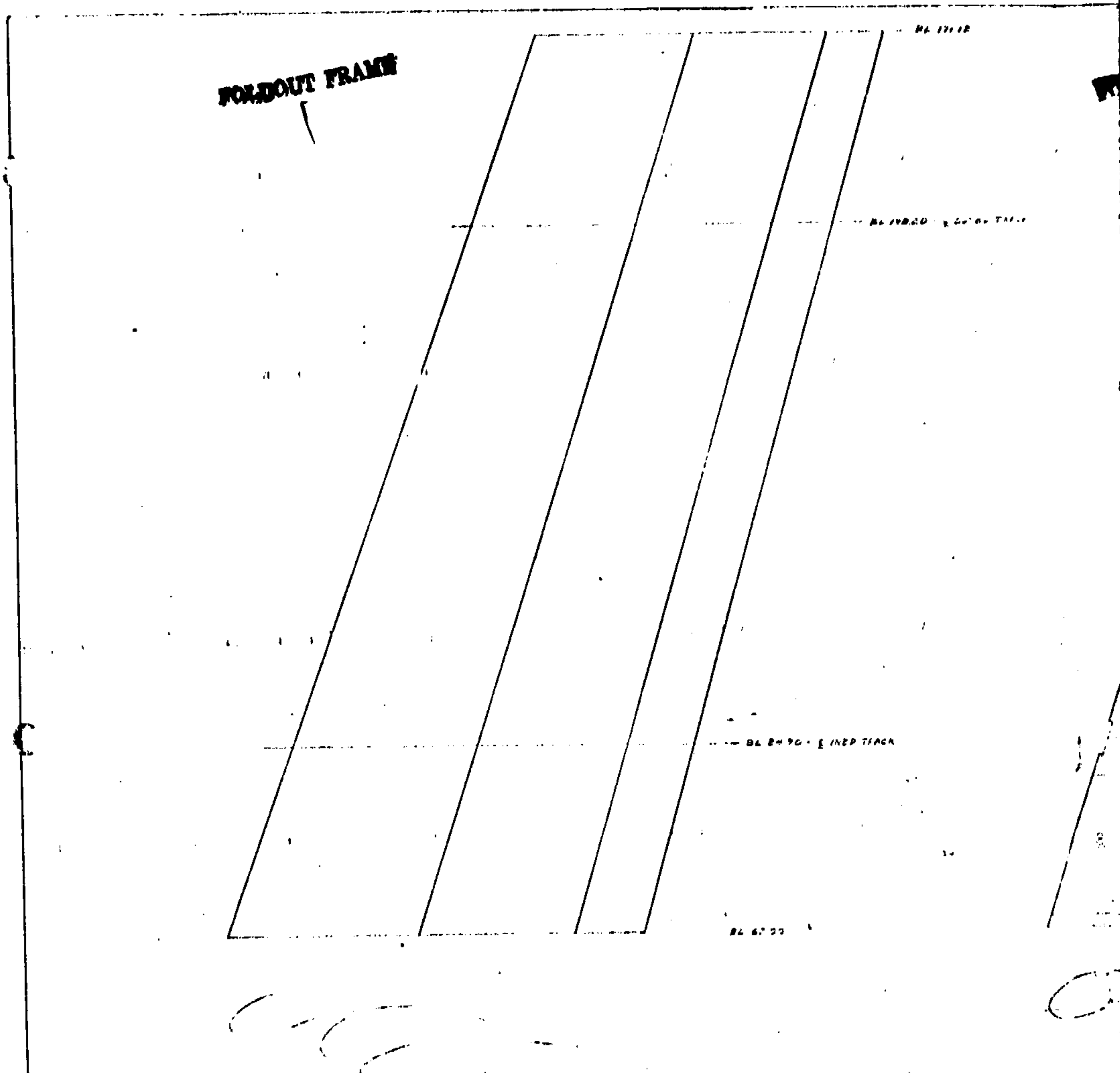


View

NOTES

STOL
STRUCTURAL ARRANGEMENT WING

FOLDOUT FRAME



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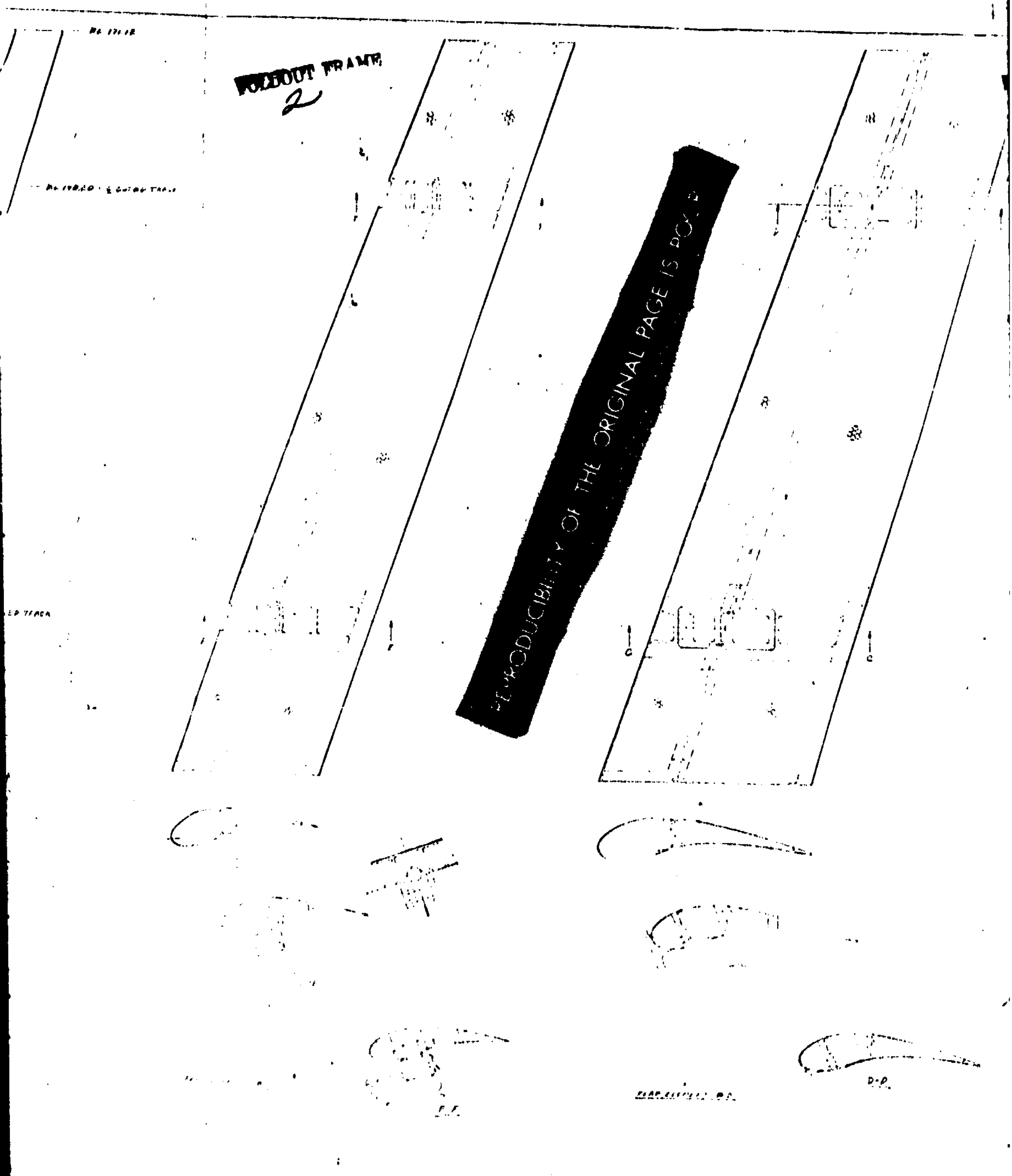

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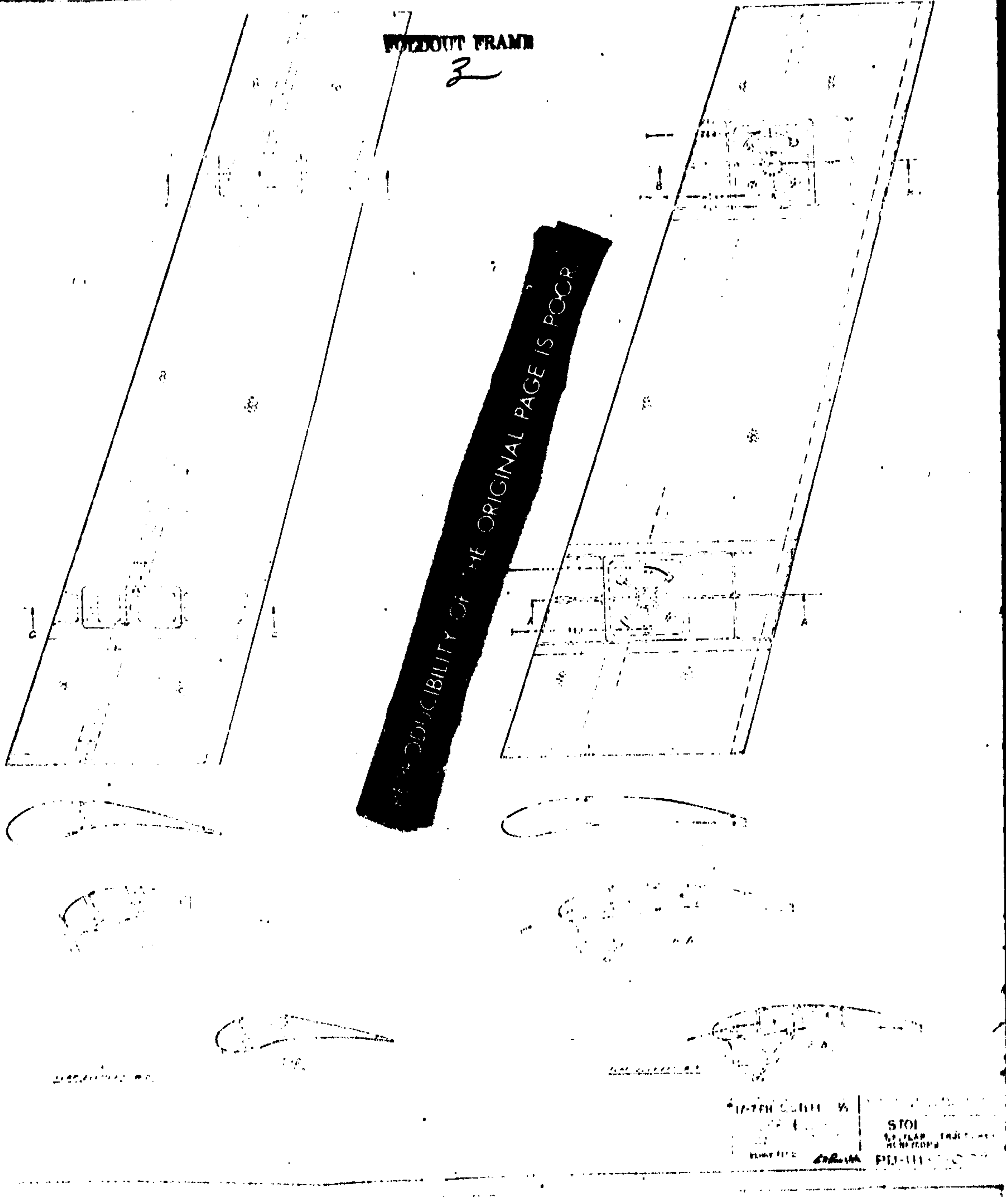

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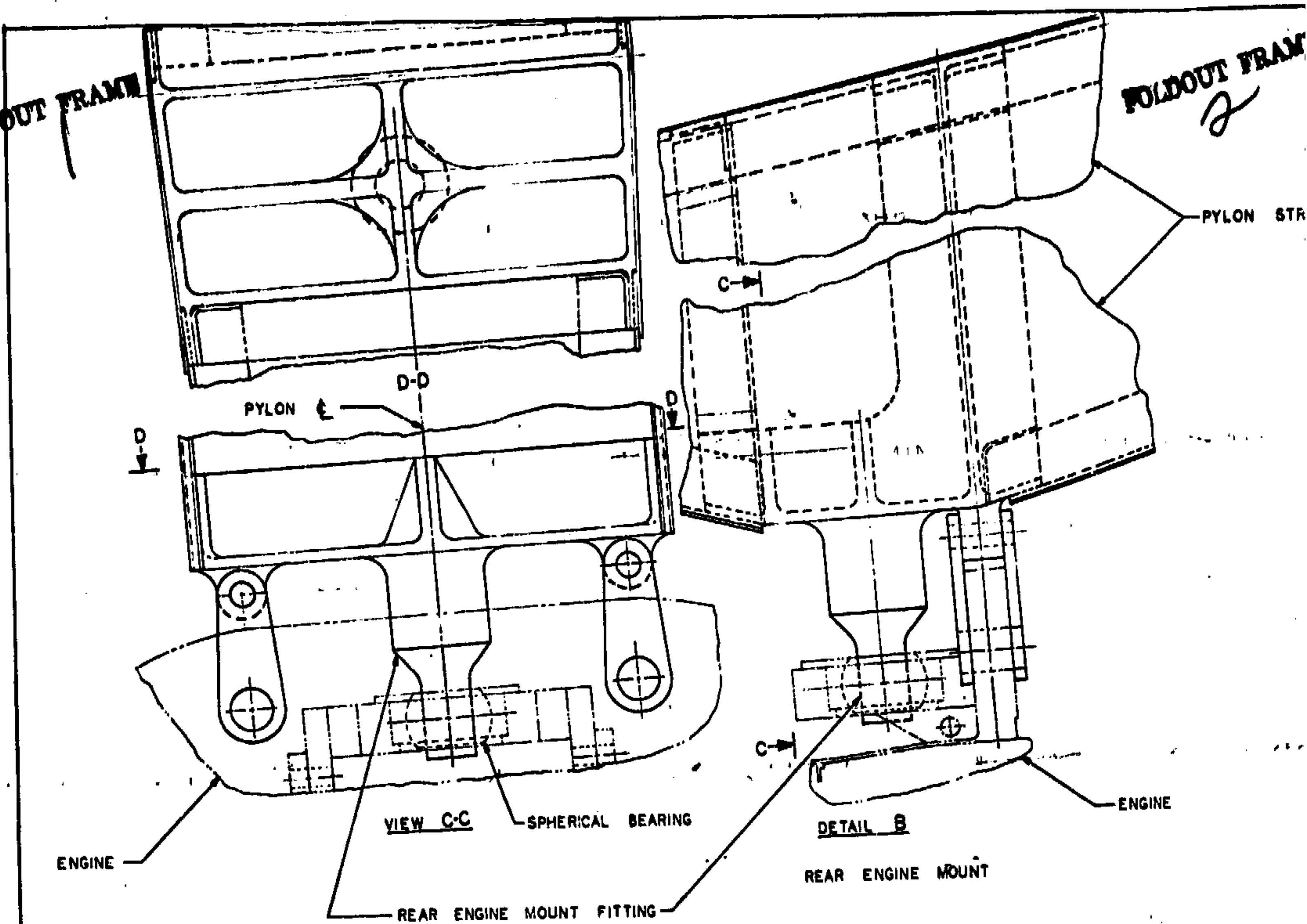
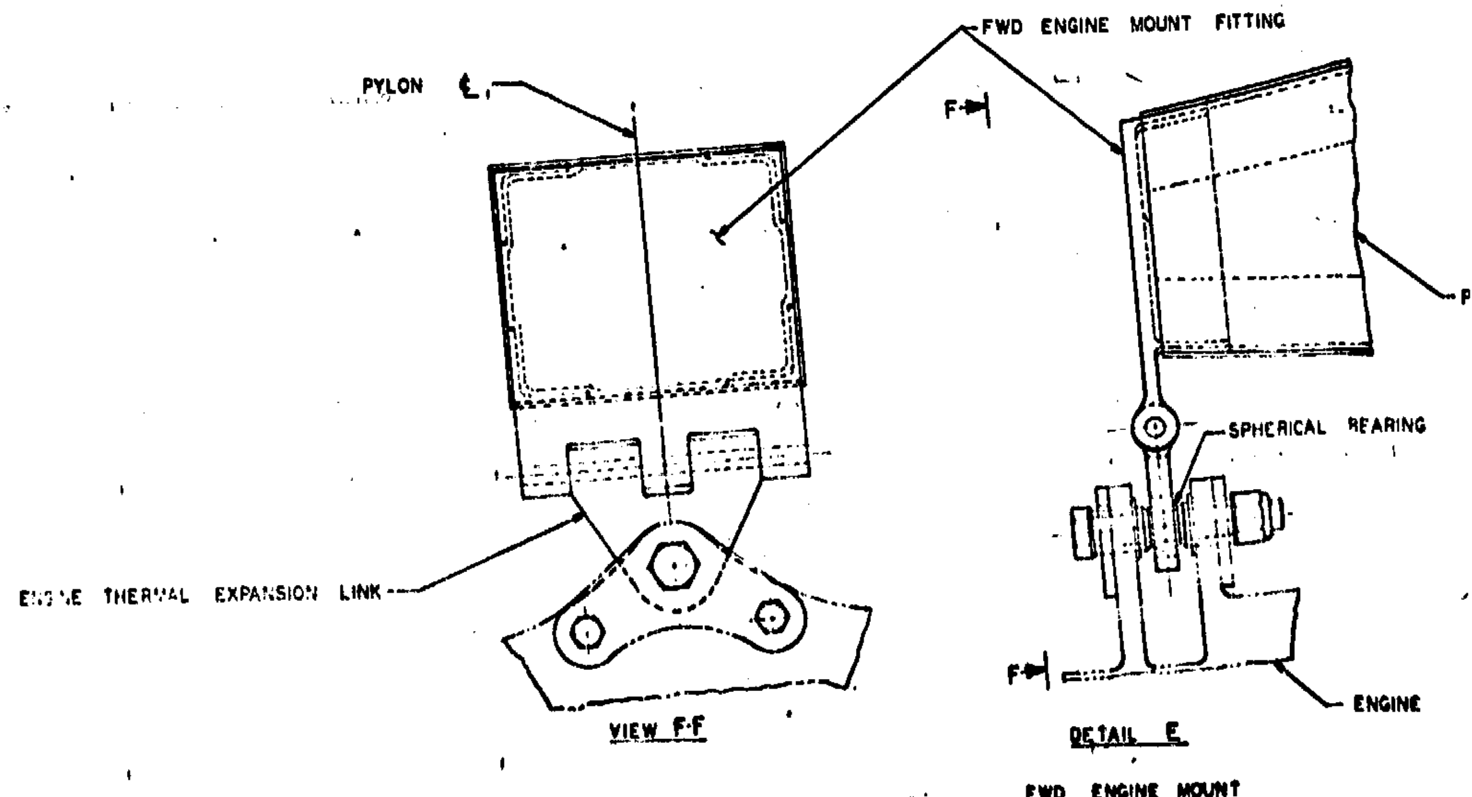
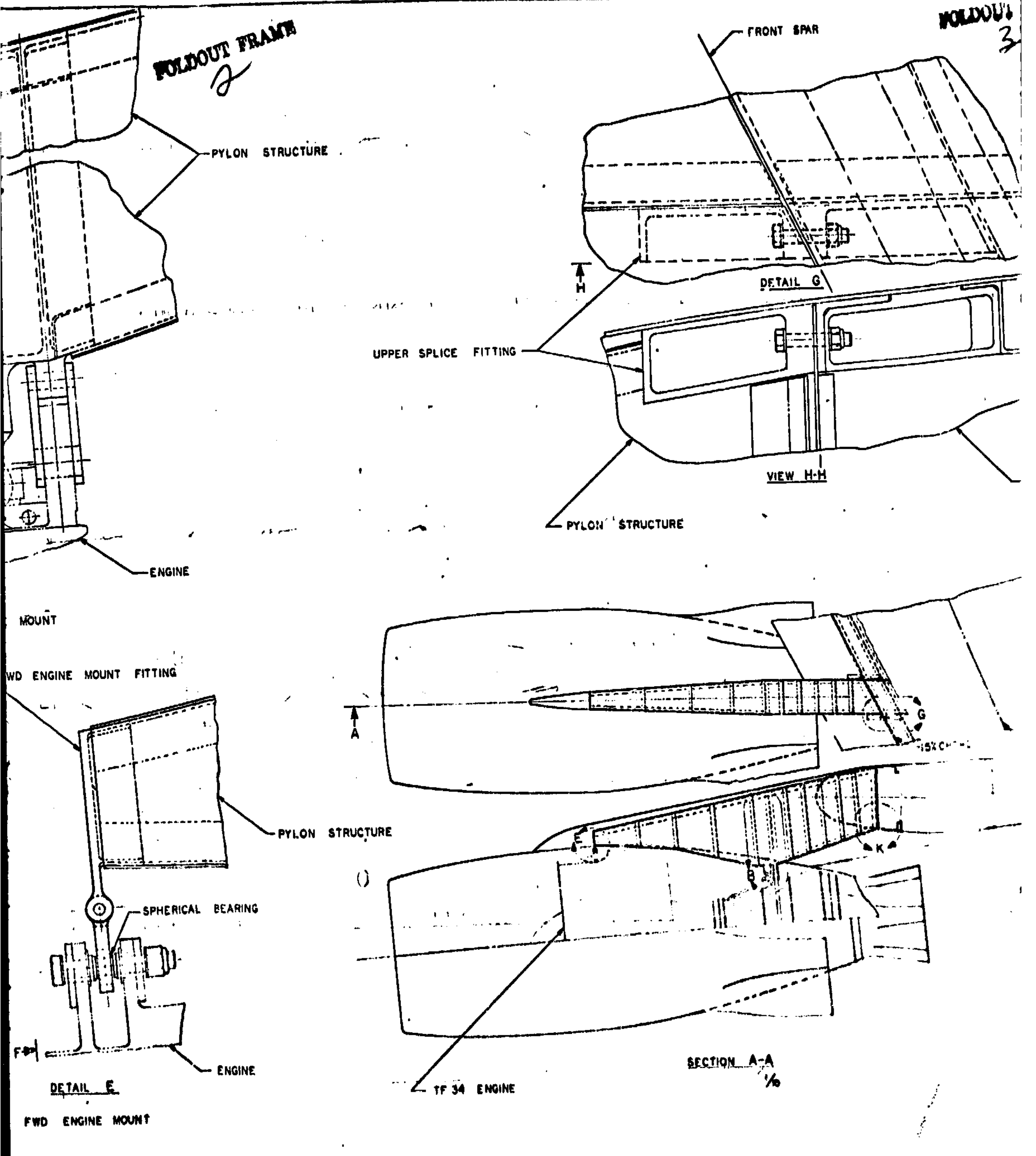
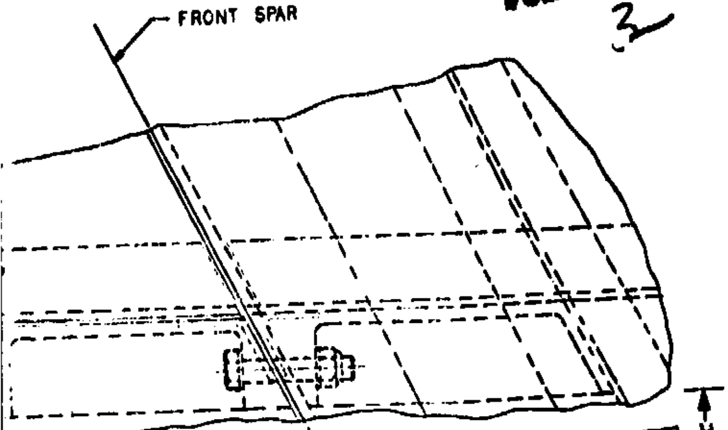




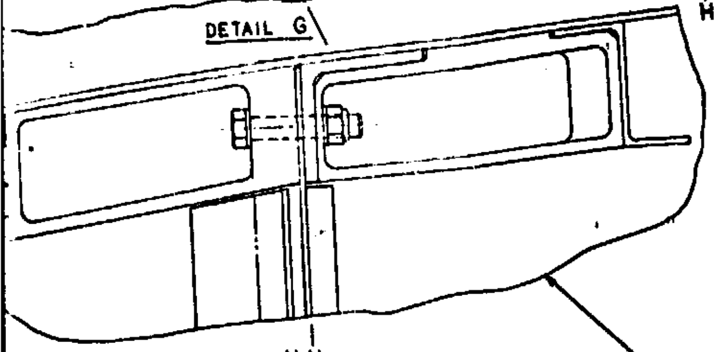

FOLDOUT FRAME 2
FOLDOUT 3
PYLON STRUCTURE
FRONT SPAR
DETAIL G
H
UPPER SPLICE FITTING
VIEW H-H
PYLON STRUCTURE
ENGINE
MOUNT
FWD ENGINE MOUNT FITTING
PYLON STRUCTURE
SPHERICAL BEARING
ENGINE
DETAIL E
FWD ENGINE MOUNT
A
G
K
TF 34 ENGINE
SECTION A-A
1/10

FOLDOUT FRAME 3

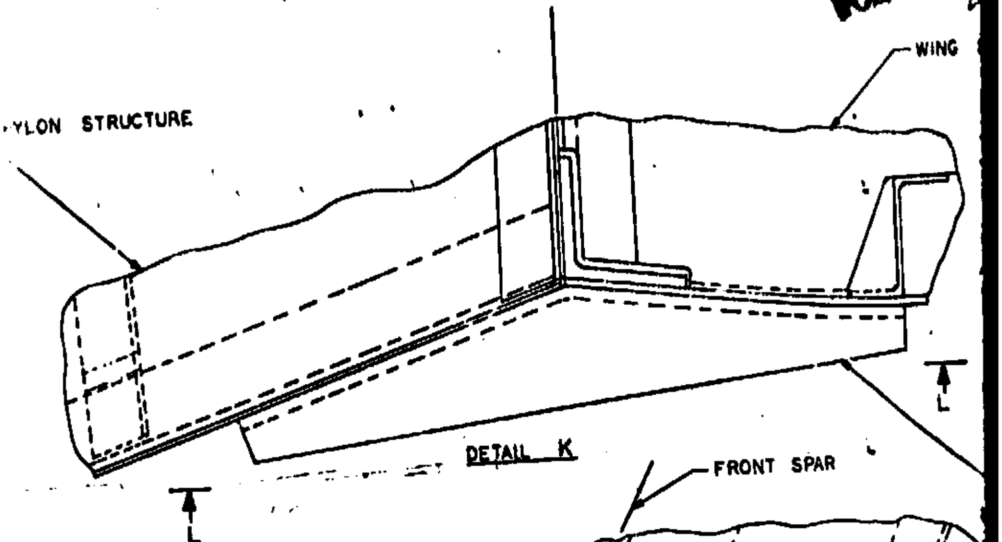
FOLDOUT FRAME



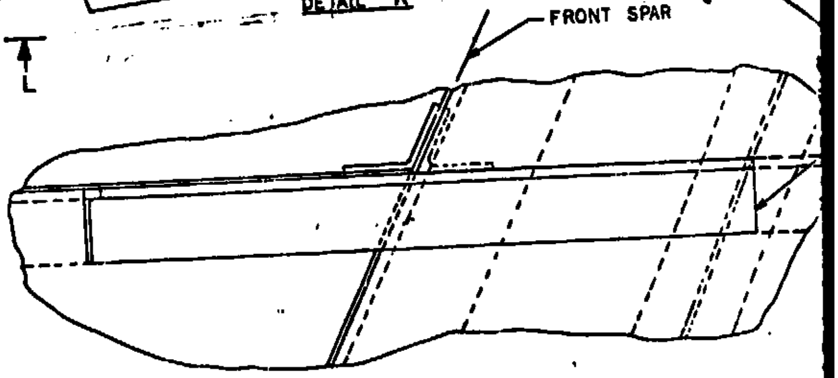
DETAIL G



VIEW H-H

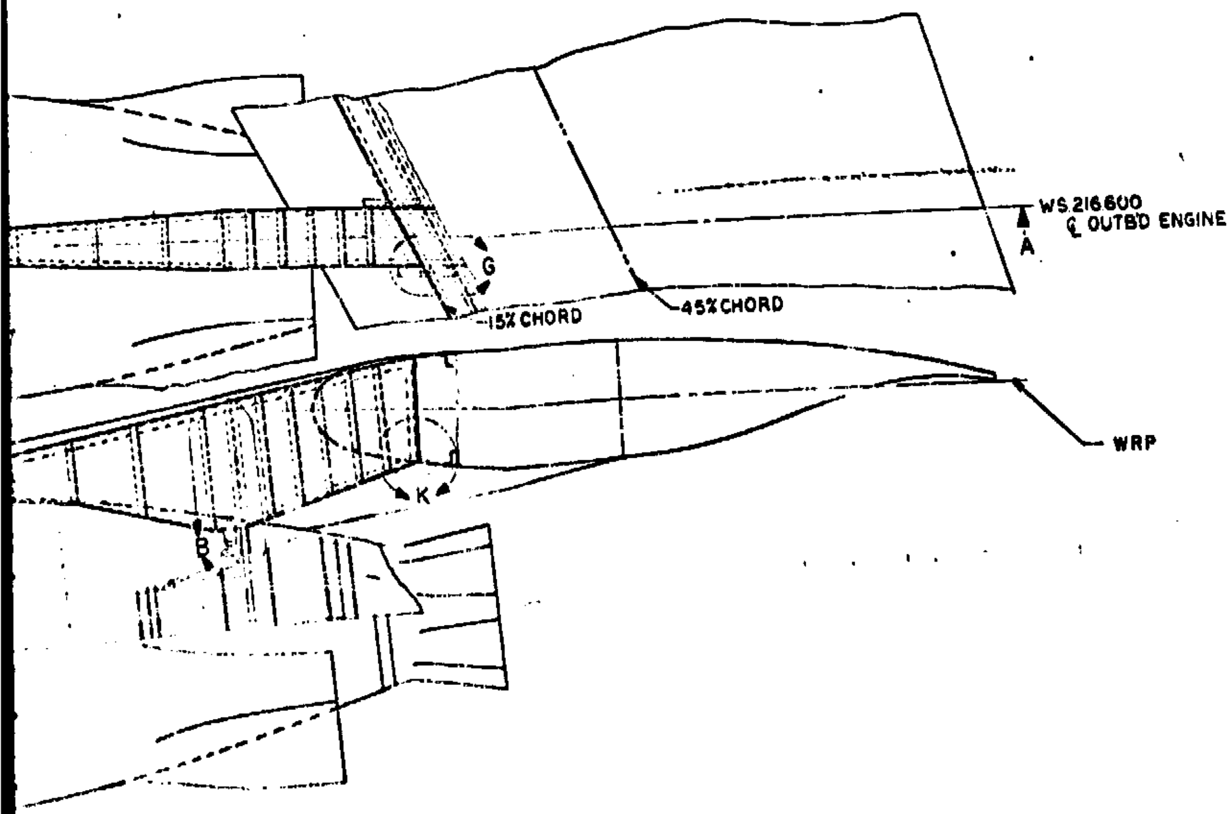


DETAIL K



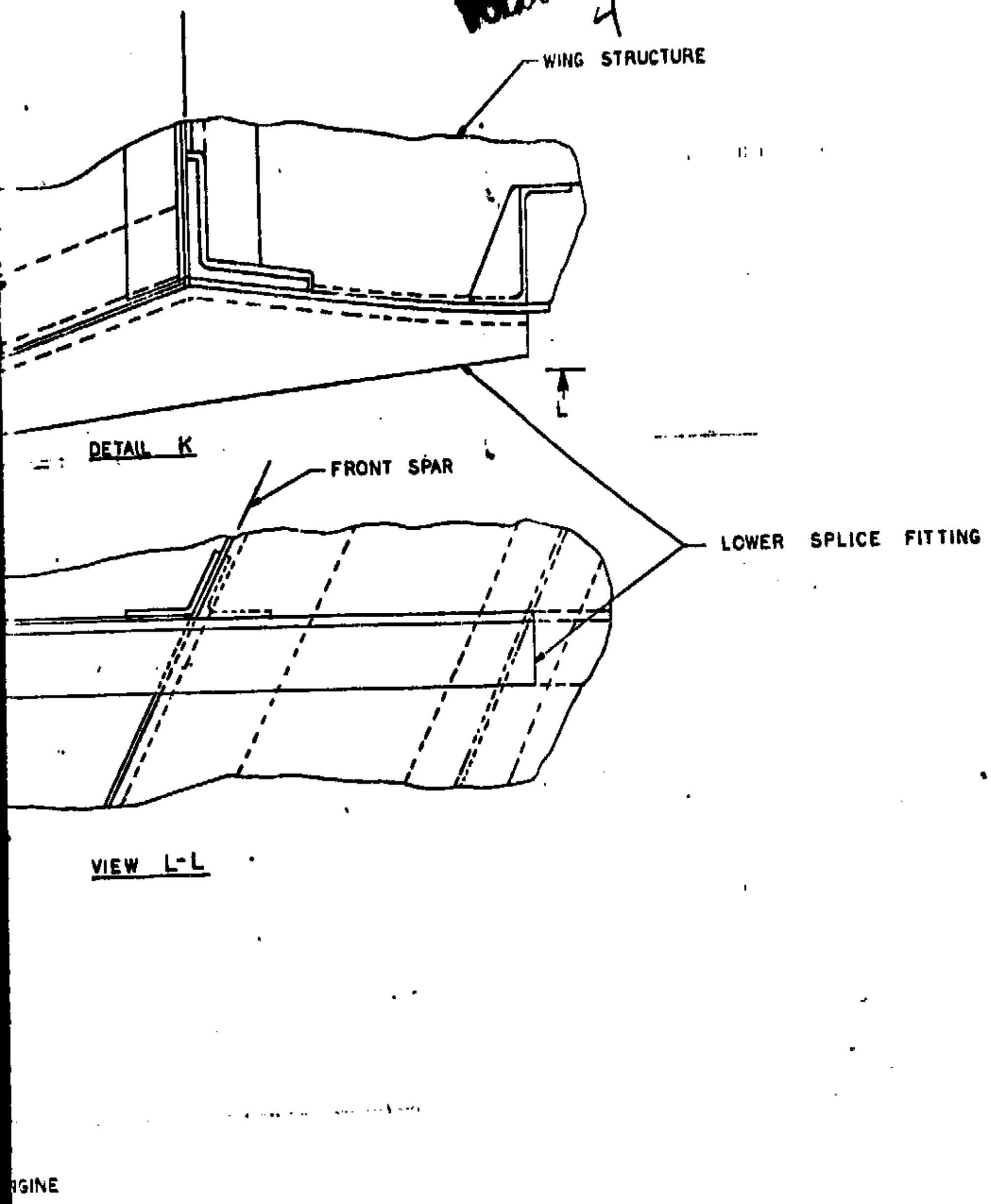
VIEW L-L

UCTURE



SECTION A-A
1/10

FOLDOUT 4



STOL
ENGINE PYLON STRUCTURE
PD-111-2-008

FOLDOUT FRAME

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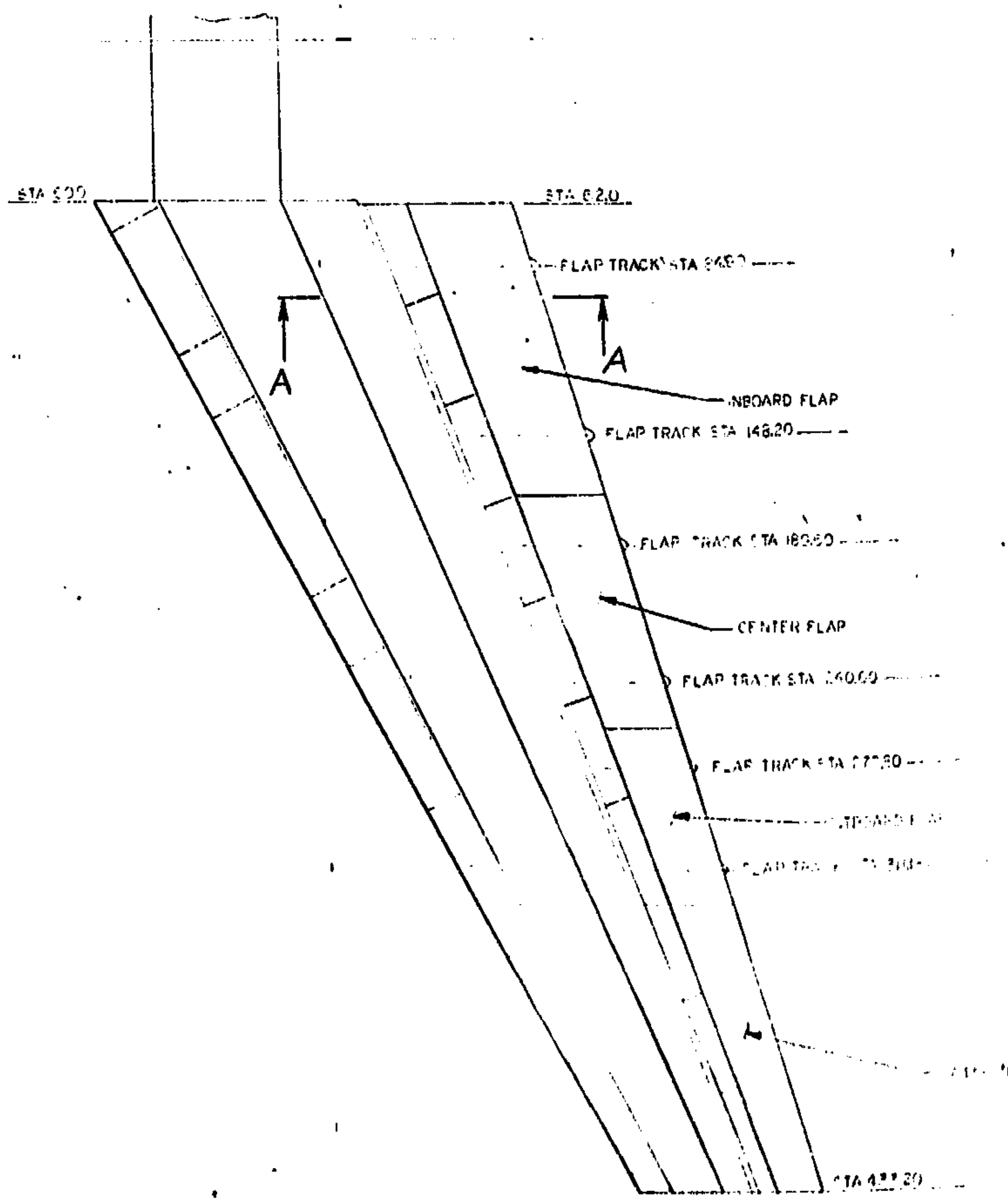

PLAN VIEW

1/20 SCALE

FOLDOUT FRAME

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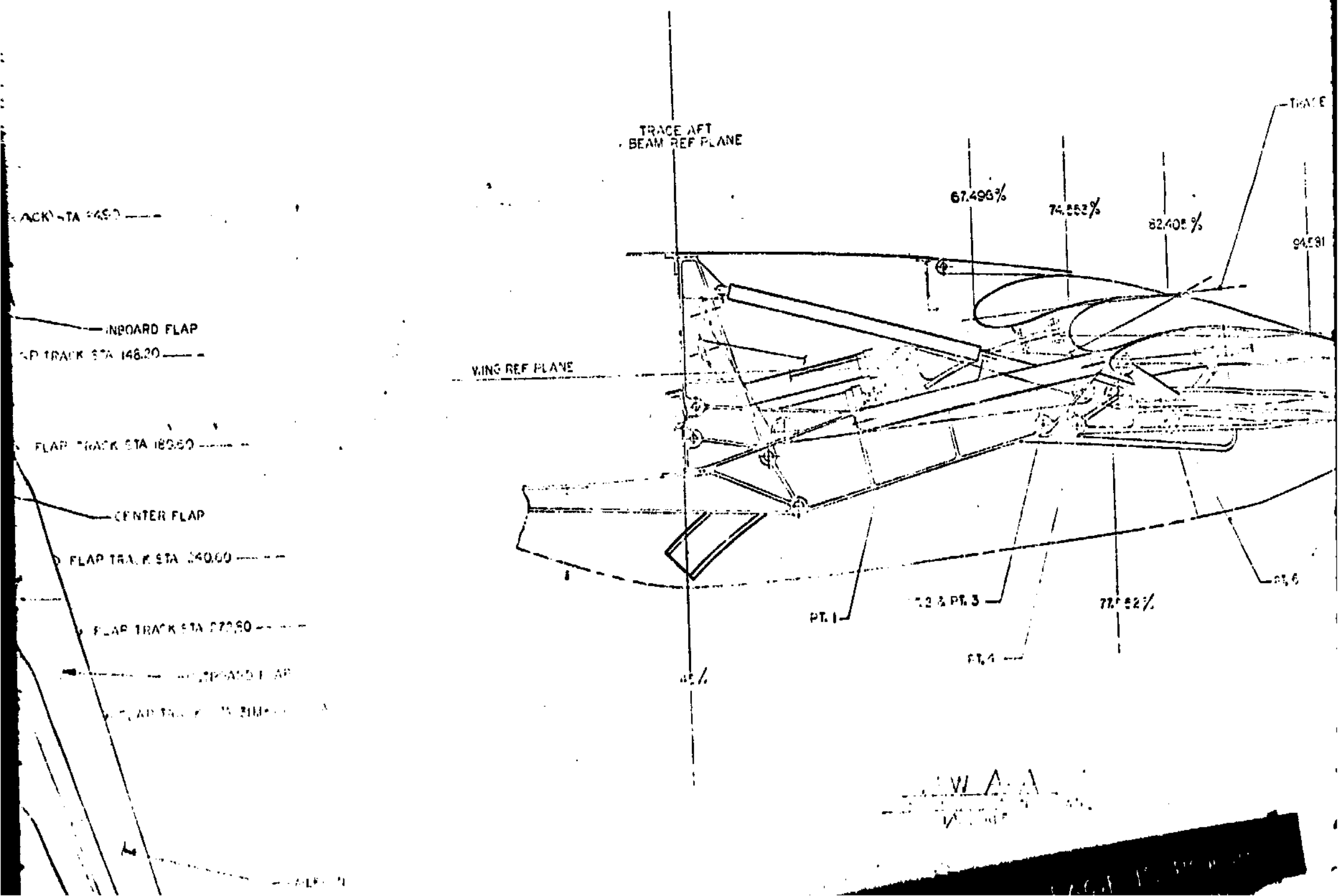

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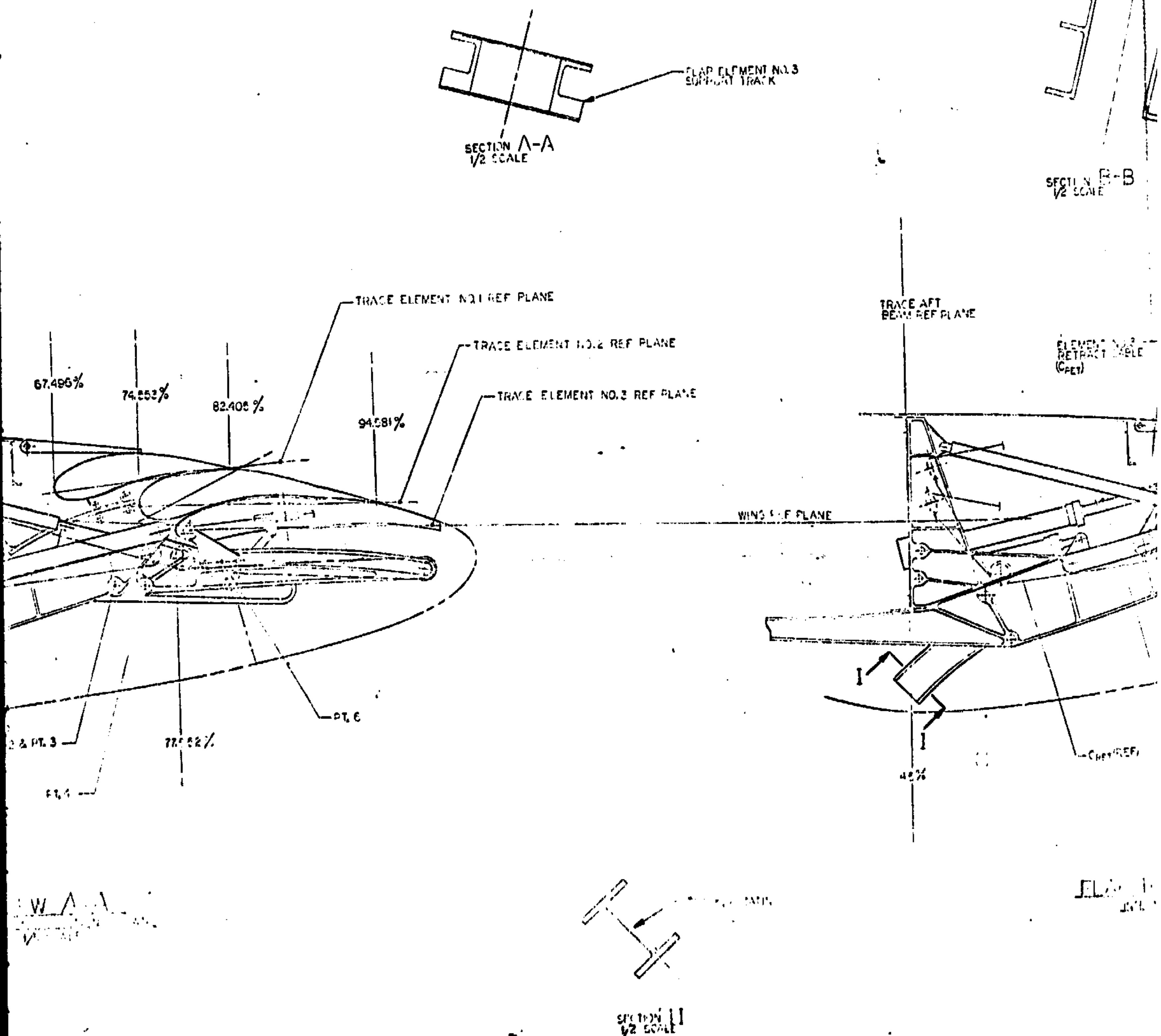

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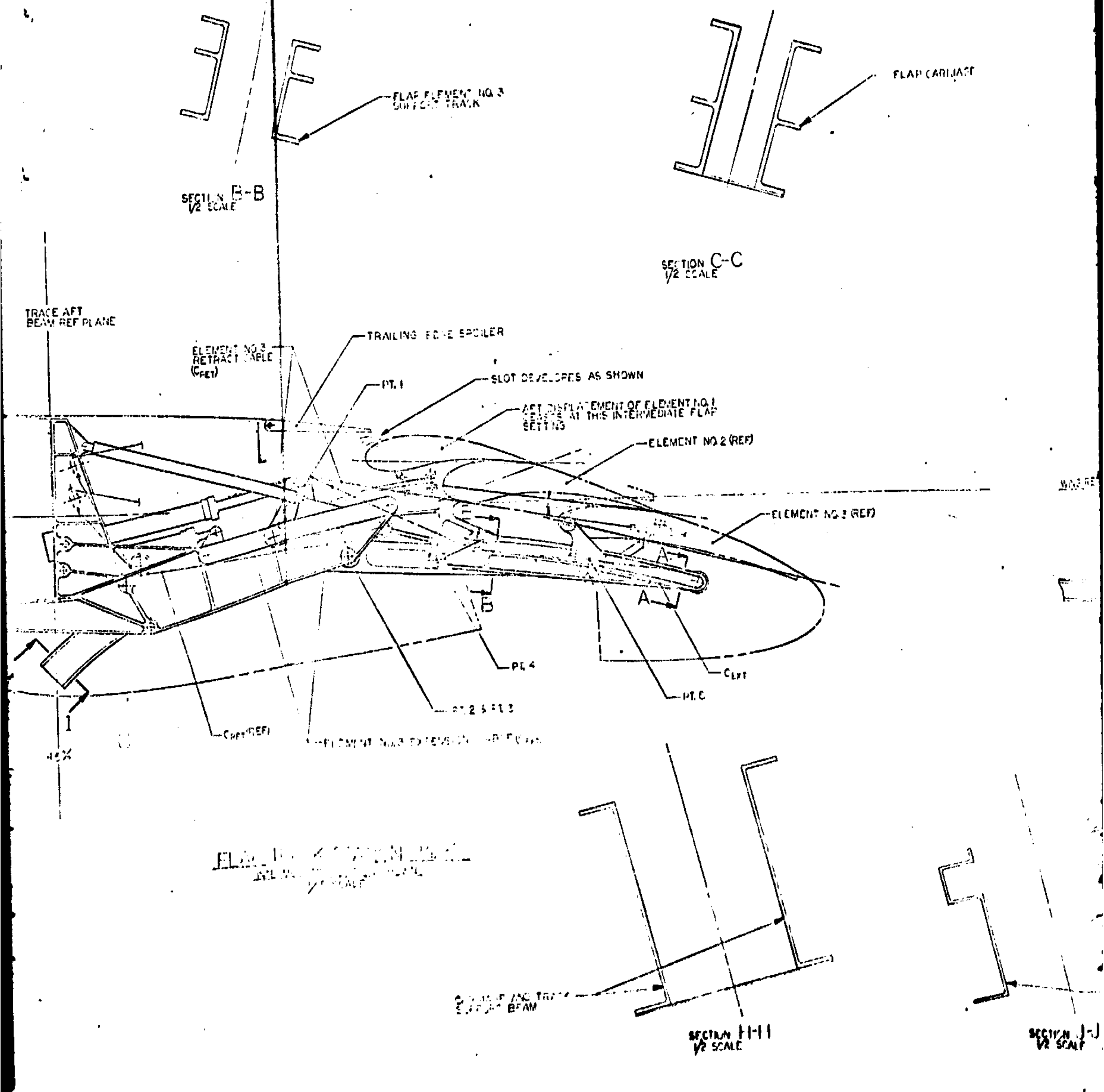

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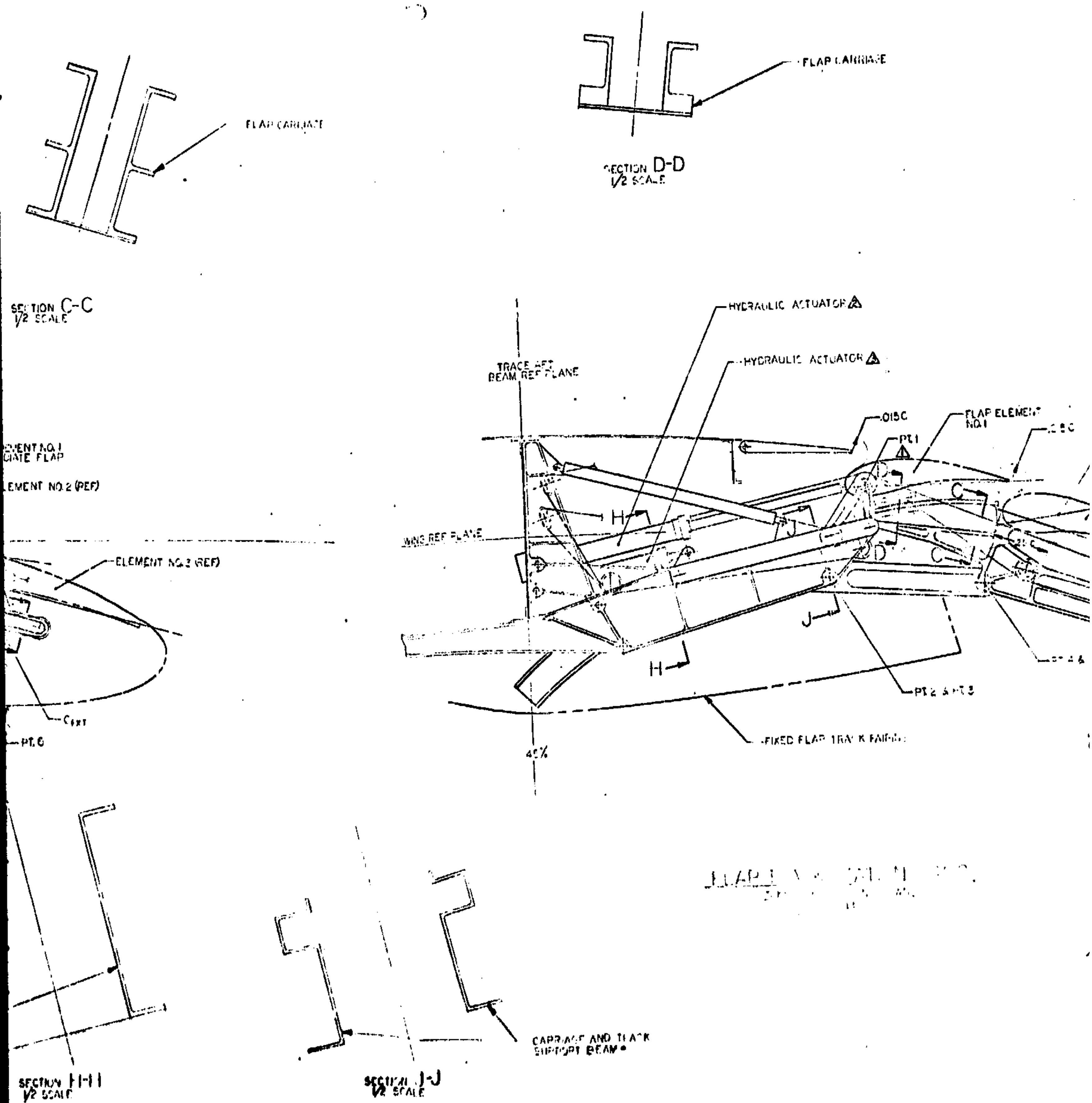

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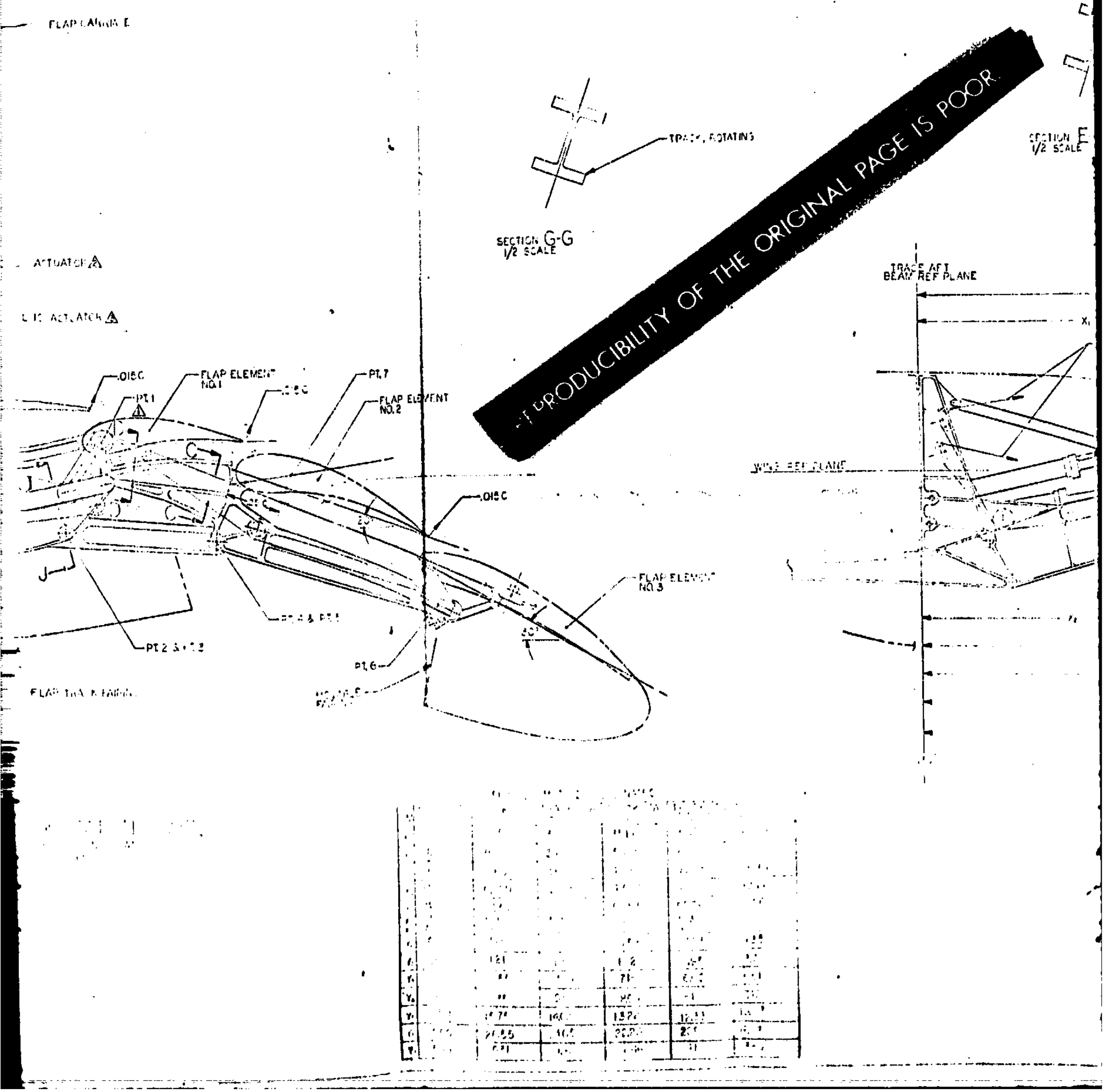
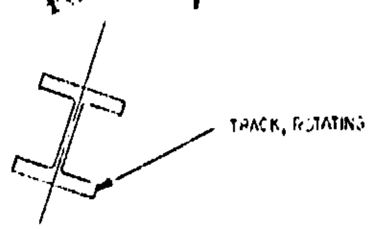
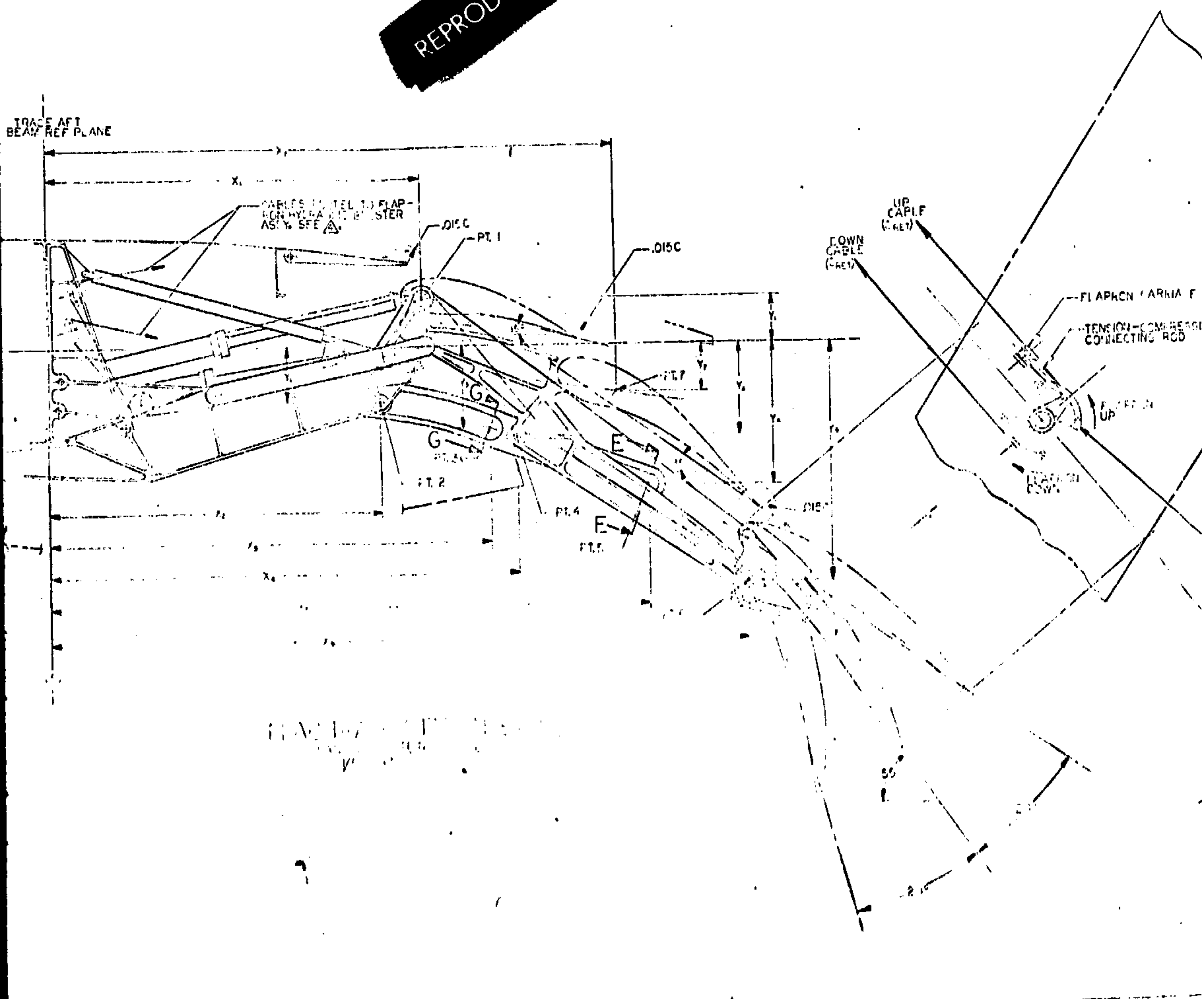




SECTION E-E
1/2 SCALE



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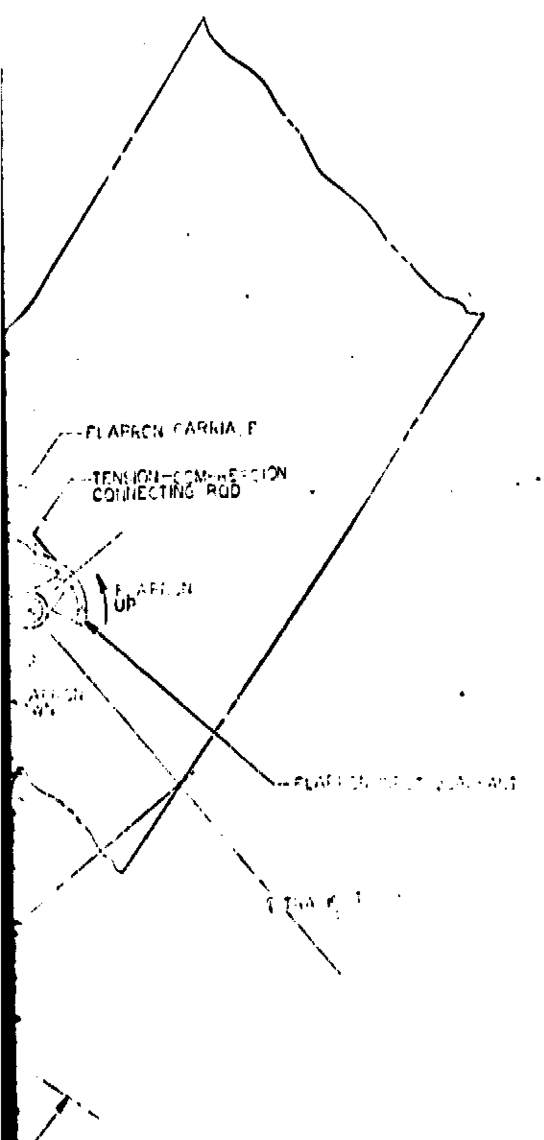

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STOL

16

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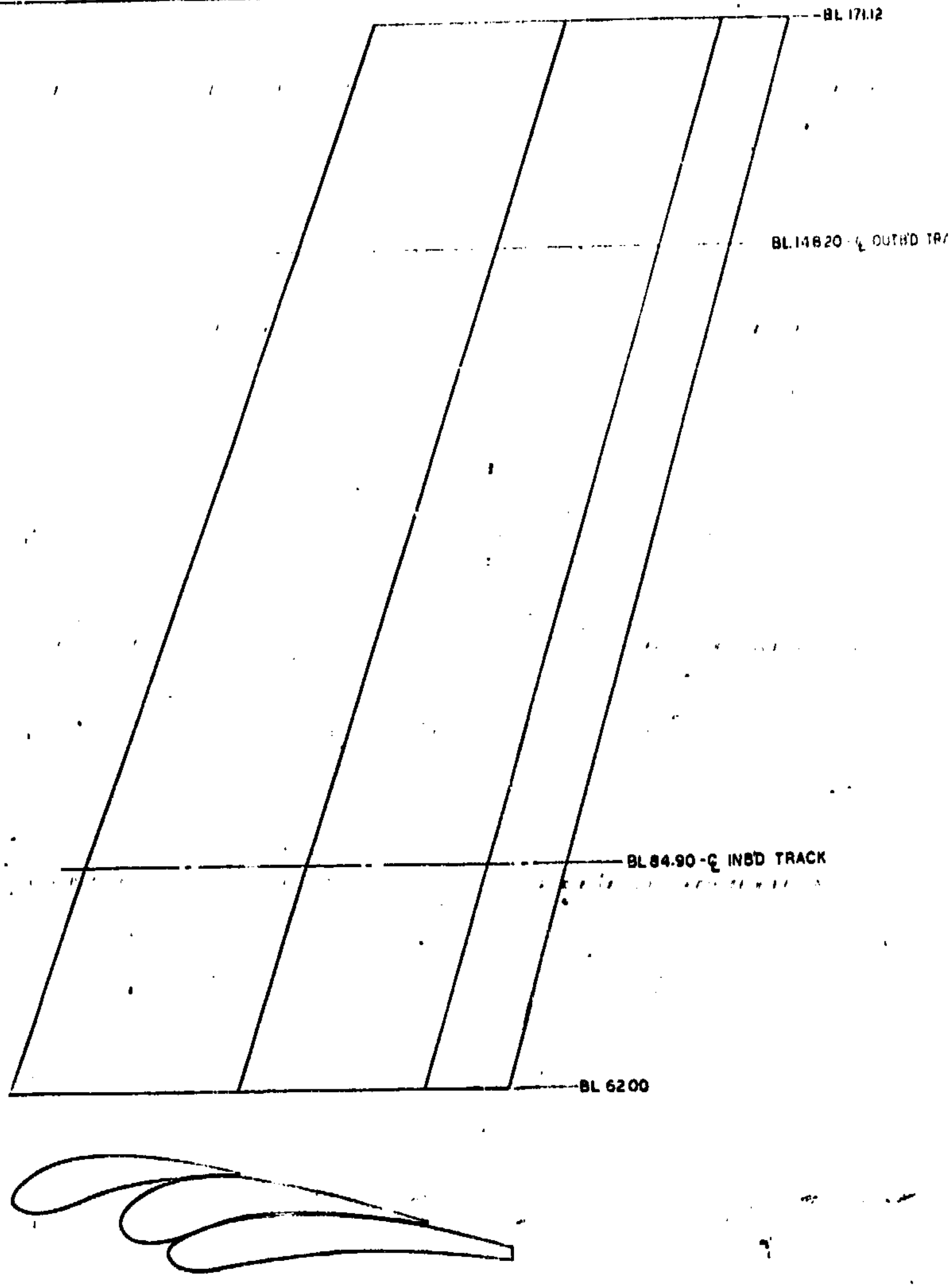
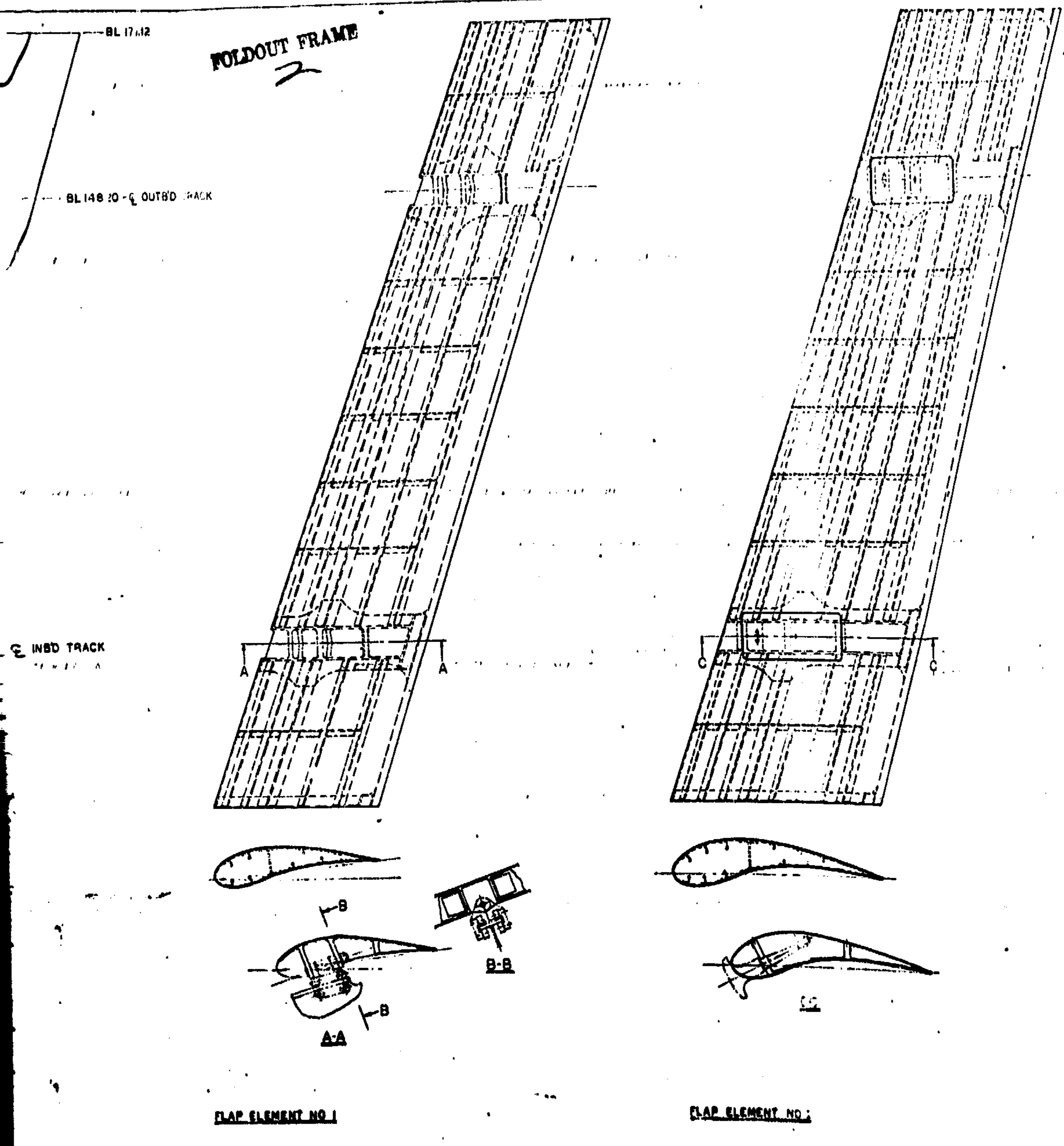


INBD FLAP-STOWED POSITION

FOLDOUT FRAME 2



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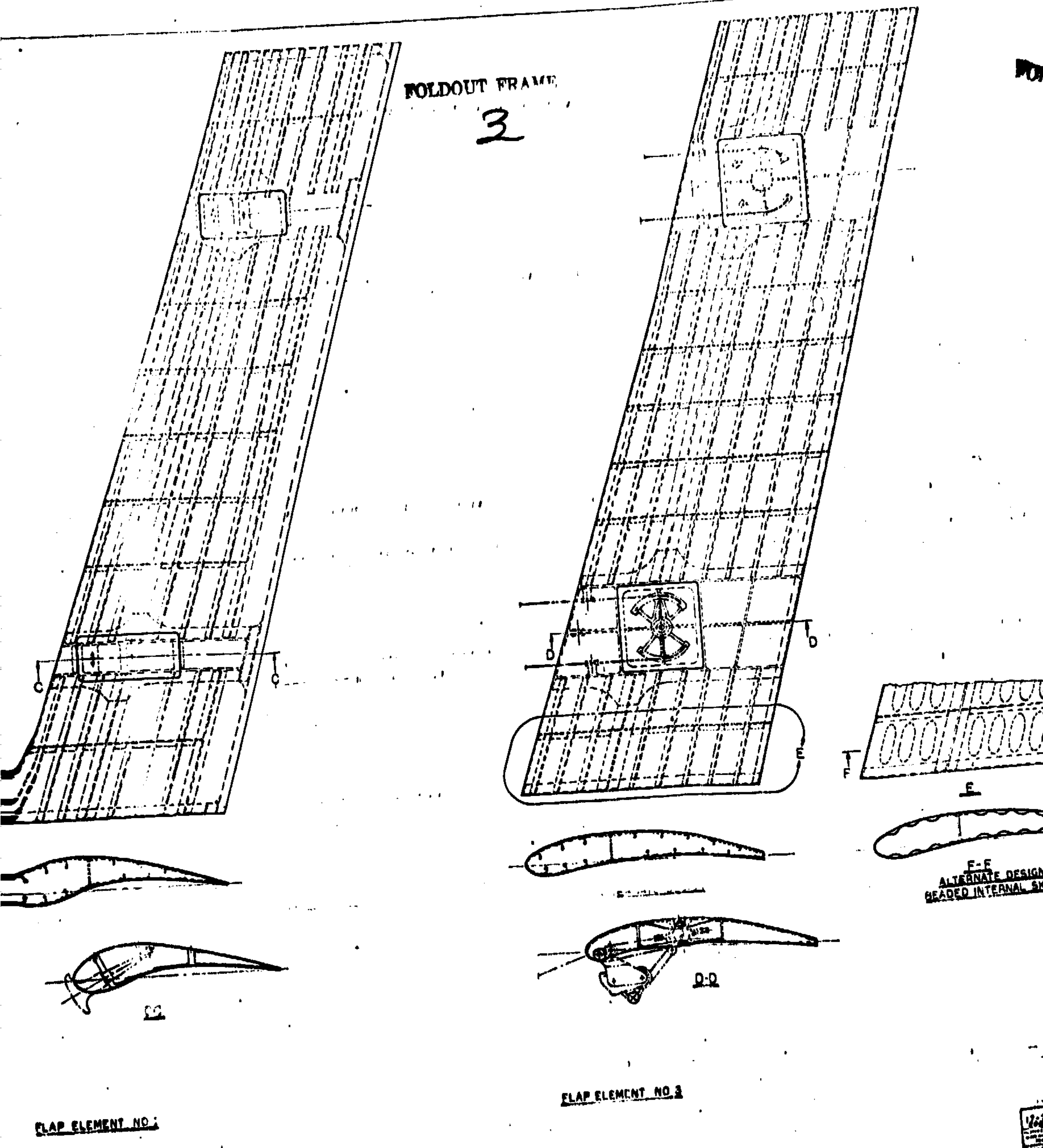

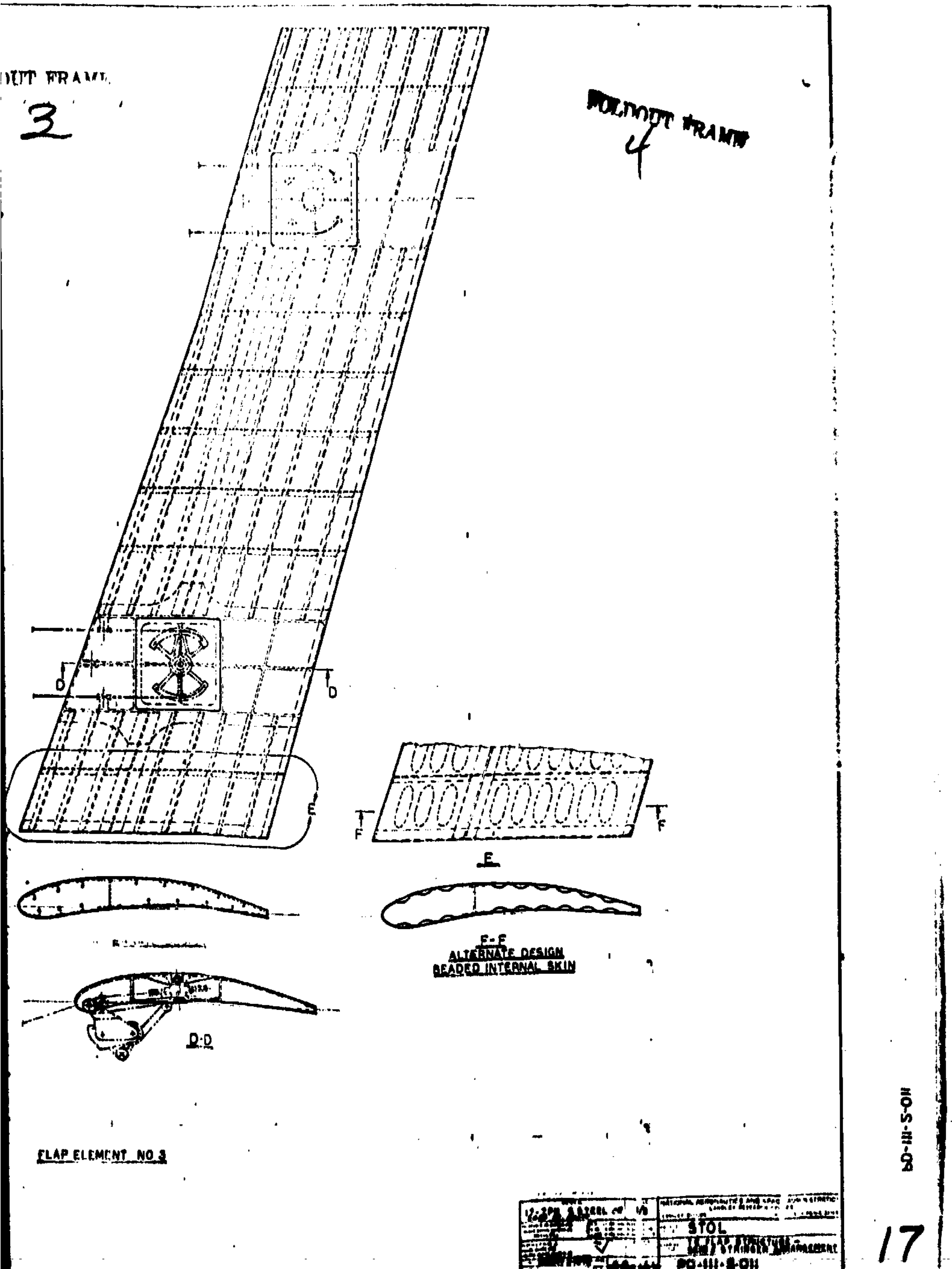

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F-F
ALTERNATE DESIGN
BEADED INTERNAL SKIN
D-D
FLAP ELEMENT NO 3
STOL
PD-111-S-011

Section II. LOADS

Static

Cruise Configuration: The wing spanwise load distribution was assumed to be represented by an elliptical distribution. The spanwise load distribution was then integrated from the tip inboard to provide the local shear and bending moment at any given spanwise station. The engines, nacelles and pylons were treated as concentrated loads. This information is represented by Figure II-1 and Figure II-2.

High Lift Configuration: The loads acting on the high lift devices were established from empirical sources at the 180 knot flight condition without blowing. In the absence of more definitive data, the spanwise distribution of section normal force and hinge moment were assumed to vary with chord length. This information is represented by Figure II-3 and Figure II-4.

The loads with the externally blown flap were investigated at a low speed flight condition for which there is limited experimental data. The results showed that local pressures on the third flap element were slightly higher with blowing than were predicted for the high speed flight condition. However, total loads were higher for all elements in the high speed flight condition.

Dynamic

Using the wing geometrical data given on Figures III-2 and III-3, and the stiffness data given on Figures III-1, III-4, III-19, III-20 and III-21, and the inertial data shown in Section IV, a finite element analysis of the wing was synthesized. This model was used to obtain, first, the natural modes of vibration and, ultimately, the flutter characteristics.

The wing was analyzed for nacelle total weights, exclusive of the mounts, of 2015 and 3500 pounds, with the nacelles both rigidly and elastically attached to the wing; it was also analyzed without any nacelles. All results are for the wing cantilevered from the fuselage side.

Vibration natural frequencies are presented in Tables II-1 to II-5. These are for the empty wing (contains only residual fuel). Not all the vibration frequencies used in the flutter analysis are given since the material would be voluminous and probably not be of primary interest.

Presented in Table II-4 are wing bending natural frequencies for the wing with nacelles elastically attached. The frequencies were obtained from an eight degree-of-freedom, lumped parameter analysis. There are six degrees-of-freedom for the wing and one degree-of-freedom for each nacelle. Mode shapes, along the wing elastic axis, that correspond to these frequencies, are shown in Figures II-5 to II-12.

Coupled bending and torsion natural frequencies, for the wing with nacelles elastically attached, are presented in Table II-5. Six control points located on the wing elastic axis with two degrees-of-freedom, one in bending and one in torsion, at each control point, were used in a dynamically coupled analysis. Again, one bending degree-of-freedom was used for each nacelle.

Results of a flutter study are presented in Tables II-6 to II-13 and in Figures II-13 and II-14. In the tables, k is the Strouhal factor and ω_F is the flutter frequency. All flutter speeds are computed in terms of true airspeeds.

The flutter analysis used:

(1) Subsonic, two dimensional, incompressible, Theodorsen theoretical aerodynamics using strip theory across the wing span

(2) The dynamics of elastically suspended nacelles, with no aerodynamics on the nacelles.

Figures II-13 and II-14 show the flutter boundaries for the wing with elastically mounted nacelles. Flutter speeds are plotted versus the ratio of nacelle fundamental bending frequency to the uncoupled fundamental bending frequency of the empty wing. The points at the nacelle-to-wing bending frequency ratio of infinity, representing a rigid mounting, are points that were determined in the study. However, the dotted line represents a judgment between its two end points.

These flutter results should be compared to those obtained by different analyses and methods; they should also be corroborated or substantiated by tests of wind tunnel dynamic models. If aerodynamic parameters from wind tunnel tests are available, it is recommended that they be used in the same program used for this study.

STOL

WING SHEAR

(RIGID)

CRUISE CONFIGURATION

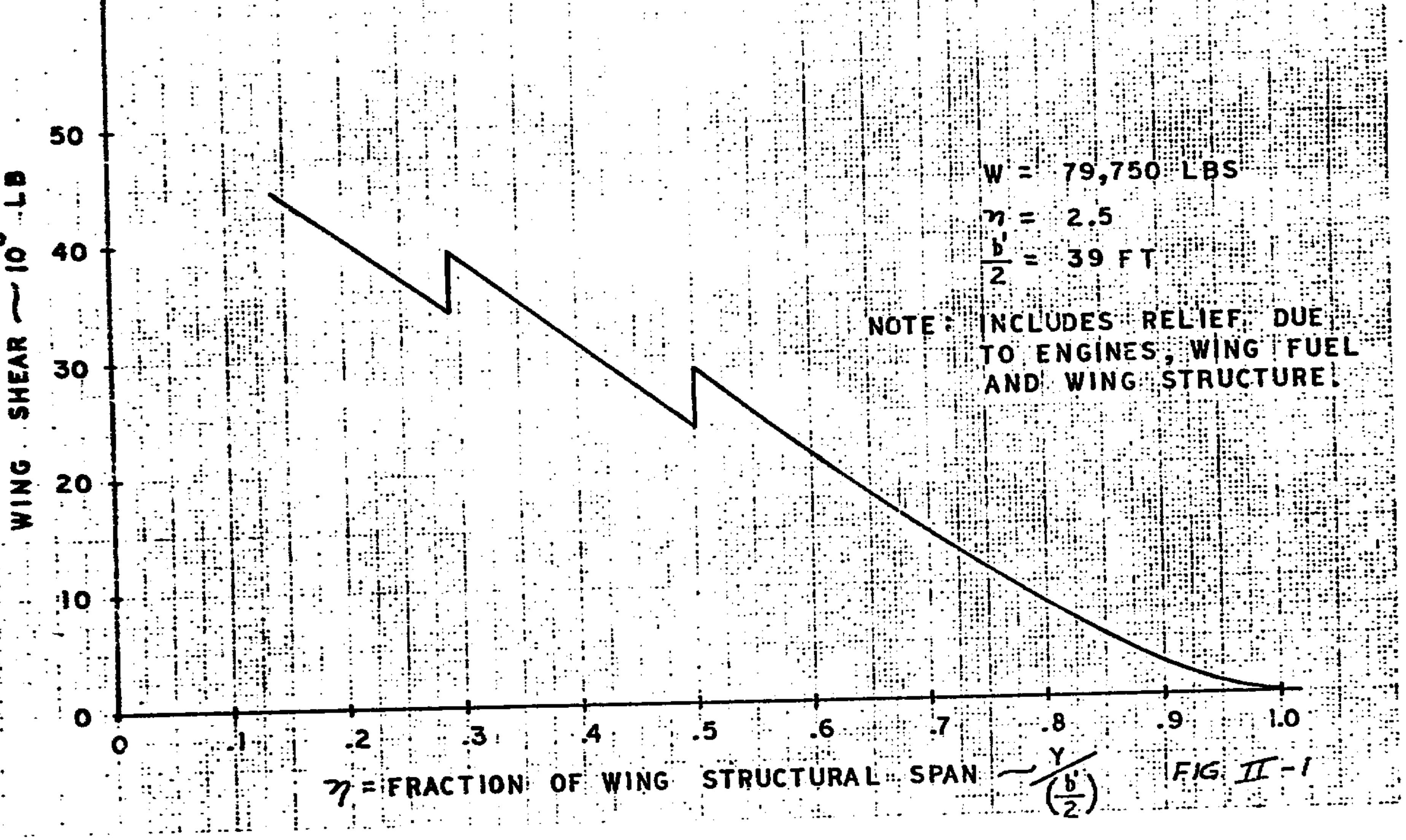


FIG II-1

STOL

WING BENDING MOMENT

(RIGID)

CRUISE CONFIGURATION

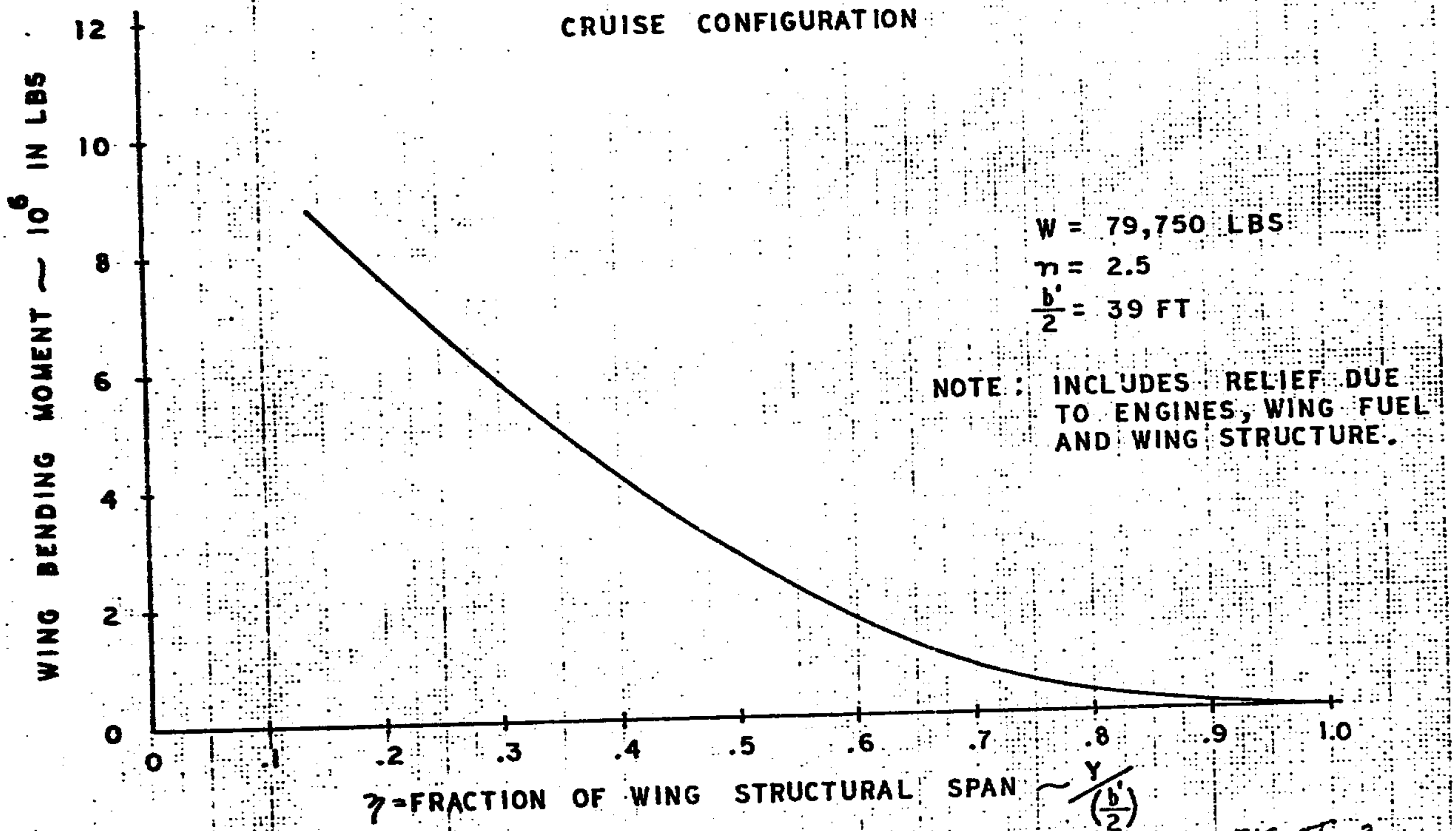


FIG. II - 2

STOL

SPANWISE DISTRIBUTION OF SECTION NORMAL FORCE (NO BLOWING)

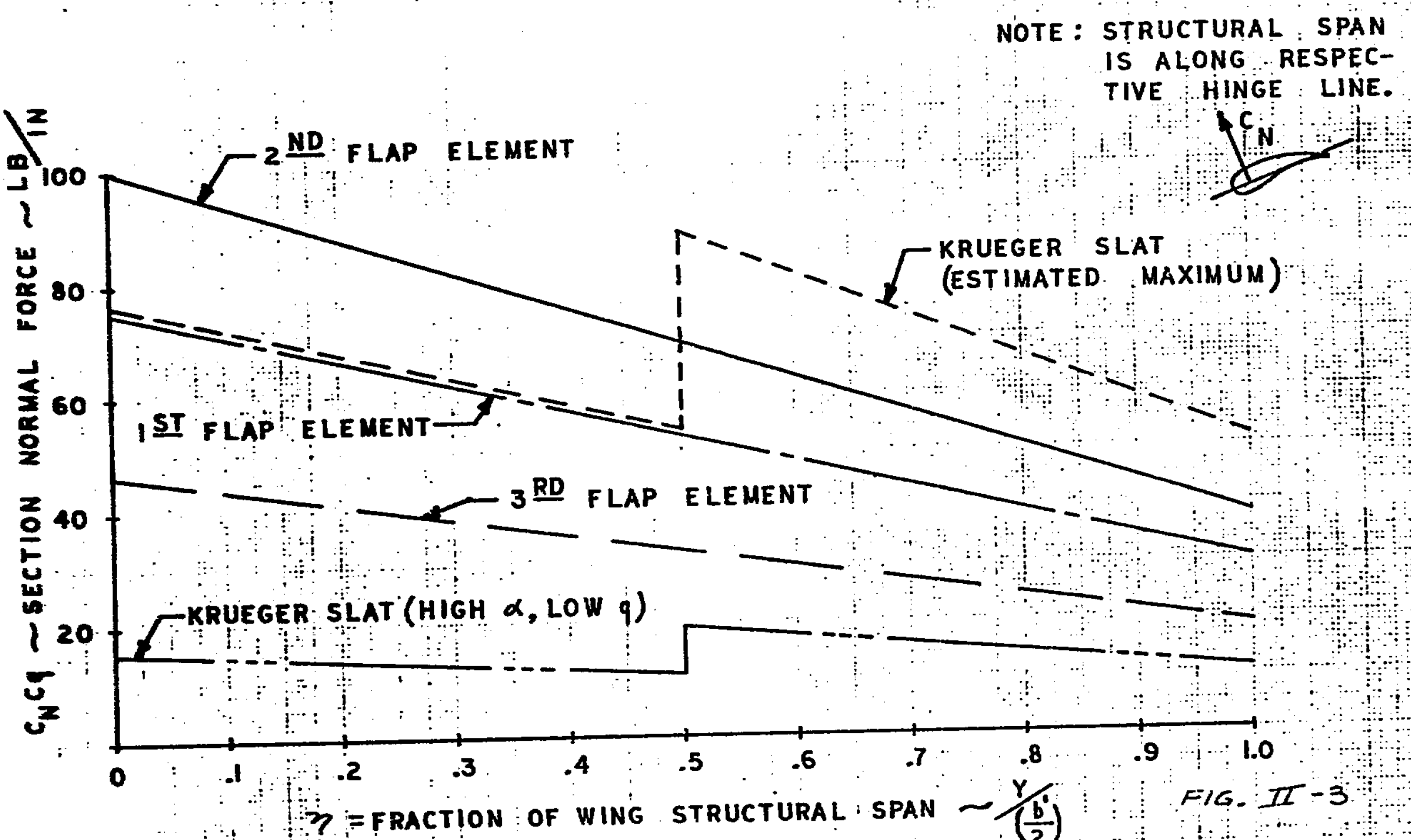


FIG. II-3

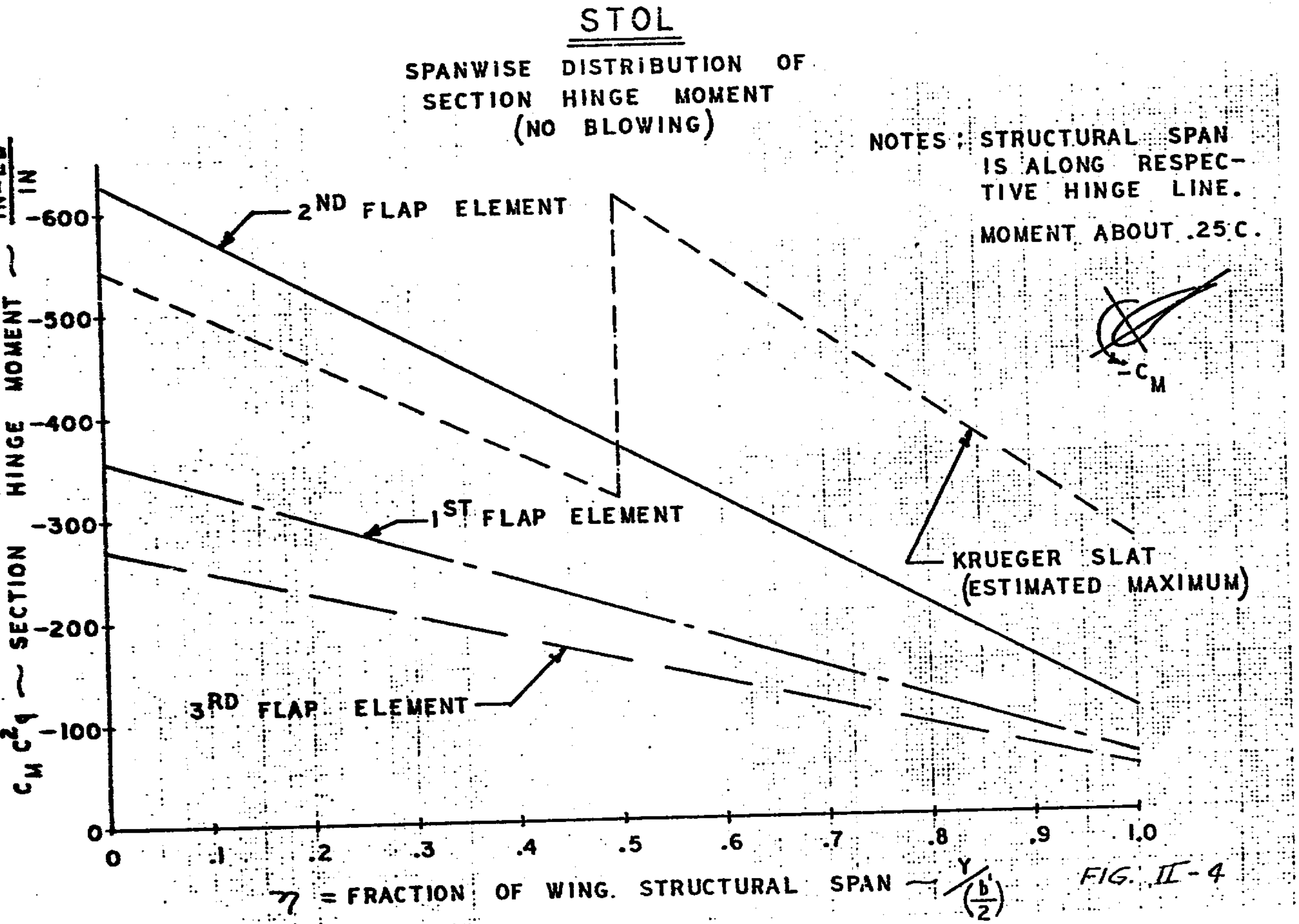

STOL
SPANWISE DISTRIBUTION OF SECTION HINGE MOMENT
(NO BLOWING)
NOTES: STRUCTURAL SPAN IS ALONG RESPECTIVE HINGE LINE.
MOMENT ABOUT .25 C.
$-C_M$
2^{ND} FLAP ELEMENT
1^{ST} FLAP ELEMENT
3^{RD} FLAP ELEMENT
KRUEGER SLAT (ESTIMATED MAXIMUM)
$C_M c^2 q$ ~ SECTION HINGE MOMENT ~ $\frac{IN\text{-}LB}{IN}$
-600
-500
-400
-300
-200
-100
0
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
η = FRACTION OF WING. STRUCTURAL SPAN ~ $Y / \left(\frac{b'}{2}\right)$

FIG. II-4

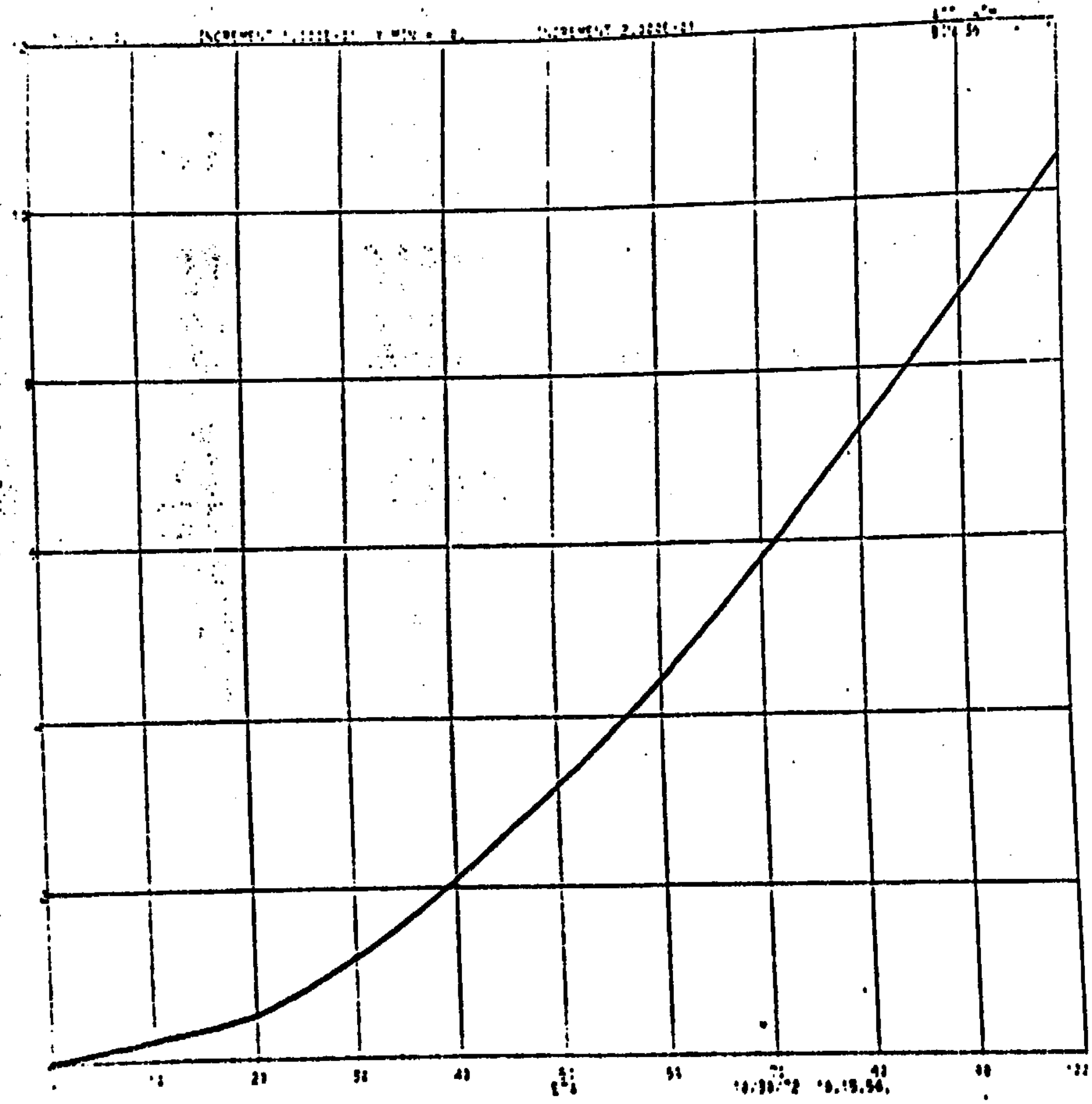


Figure II-5-Natural bending mode shape along wing elastic axis. STOL wing (empty). ω = 18.49 rad/sec.

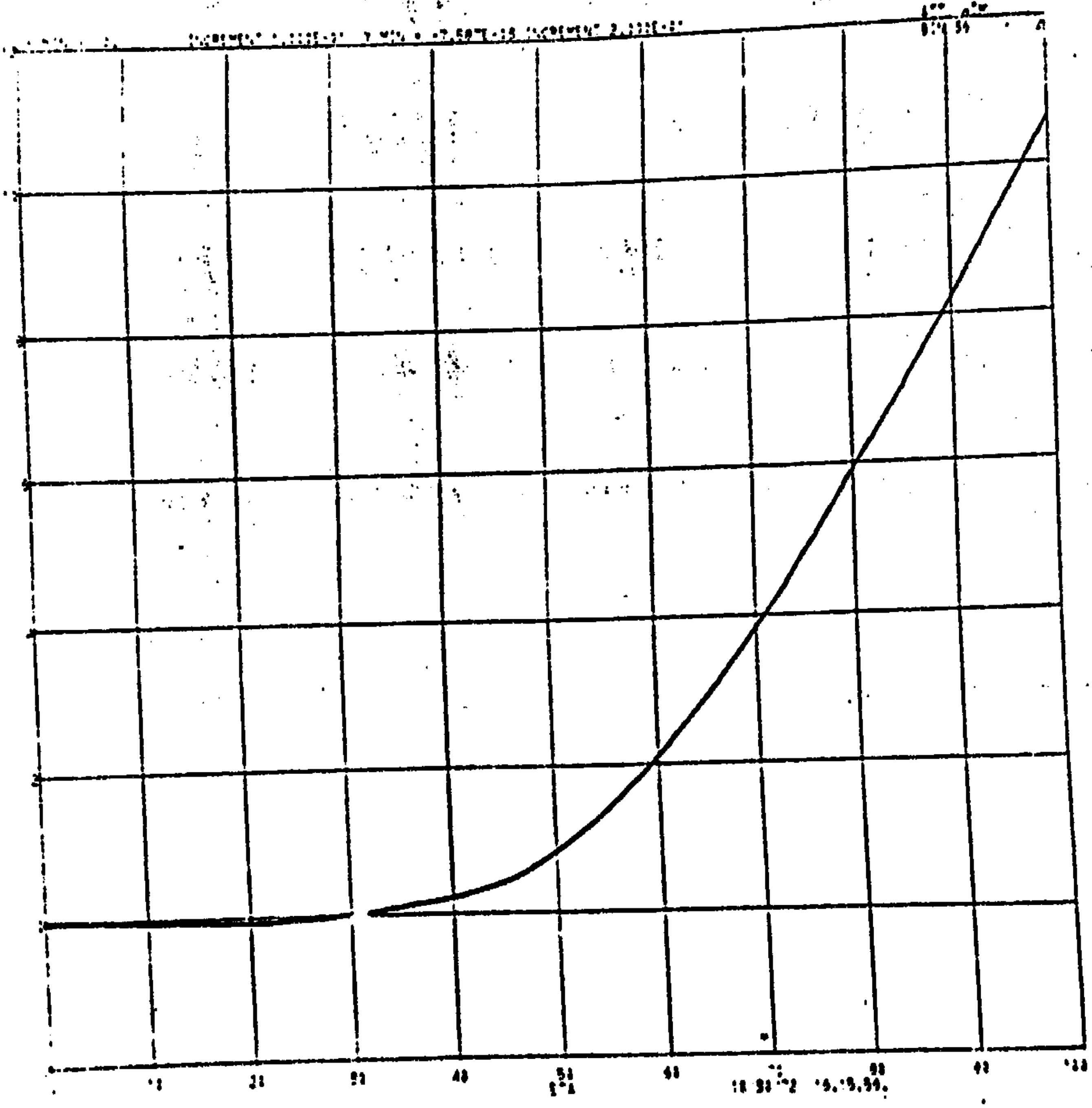


Figure II-6 Natural bending mode shape along wing elastic axis.
STOL wing (empty). ω = 35.68 rad/sec.

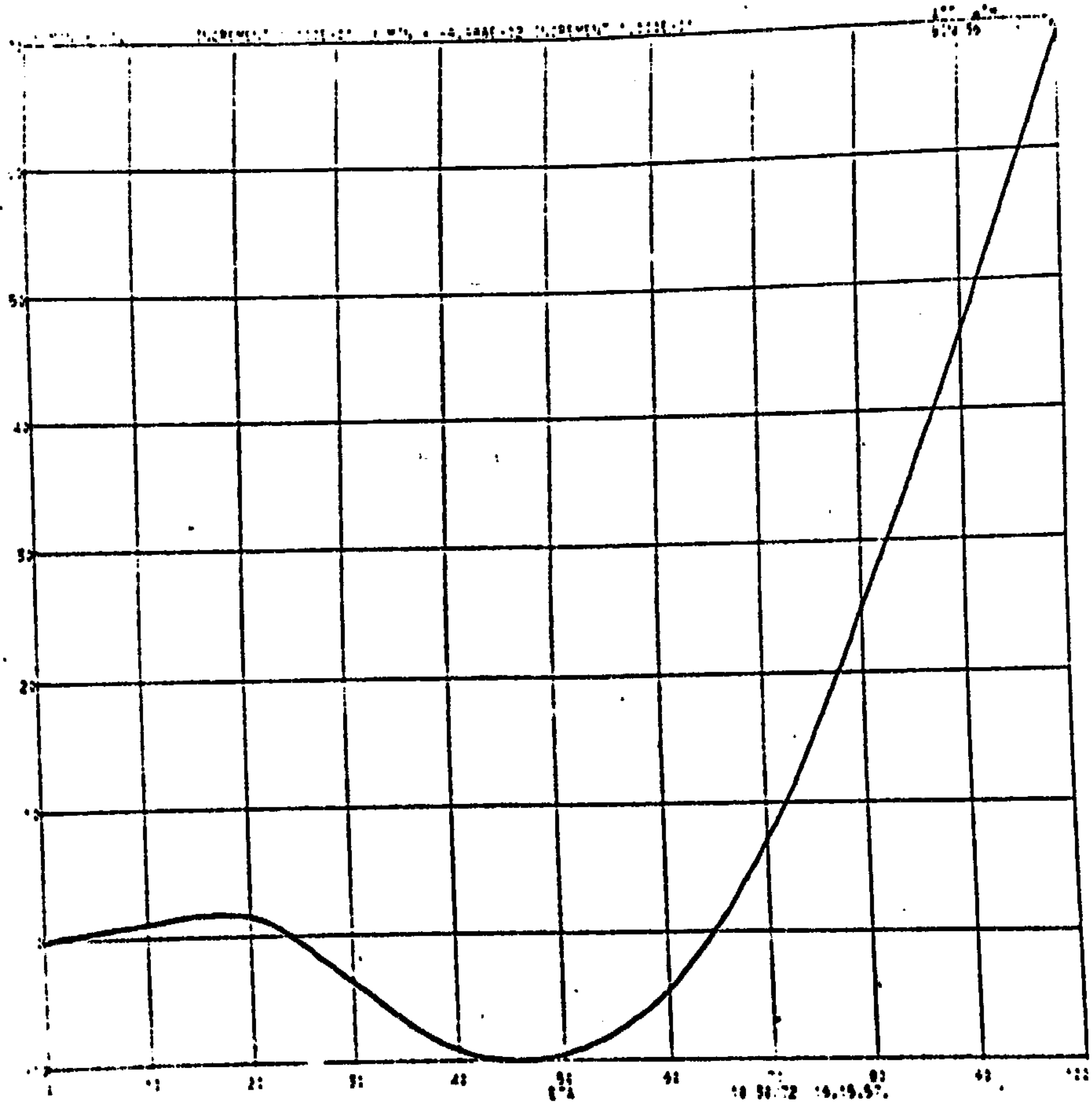


Figure II-7 Natural bending mode shape along wing elastic axis.
STOL wing (empty) ω = 58.73 rad/sec.

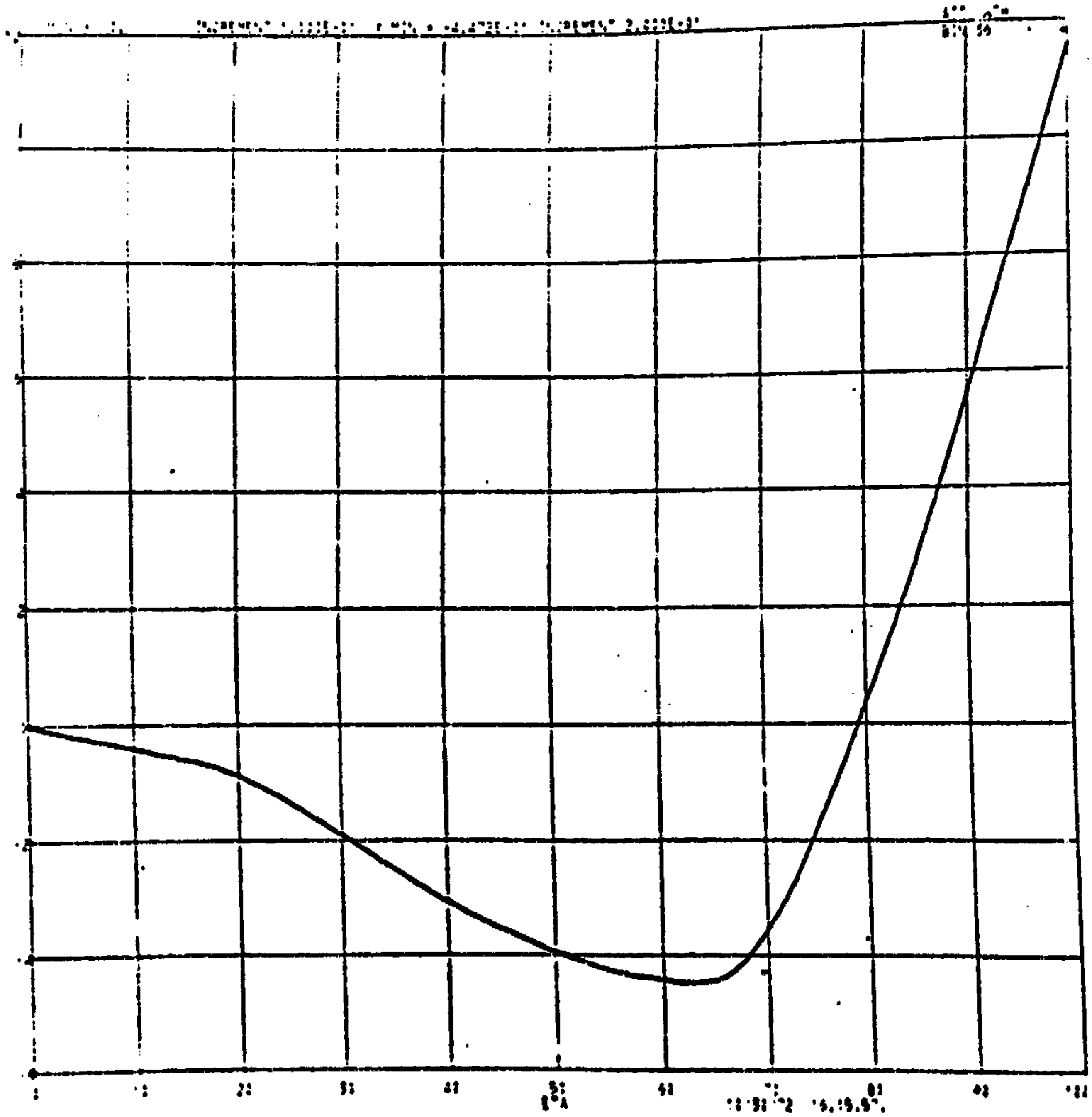


Figure II-8 Natural bending mode shape along wing elastic axis.
STOL wing (empty). ω = 104.9 rad/sec.

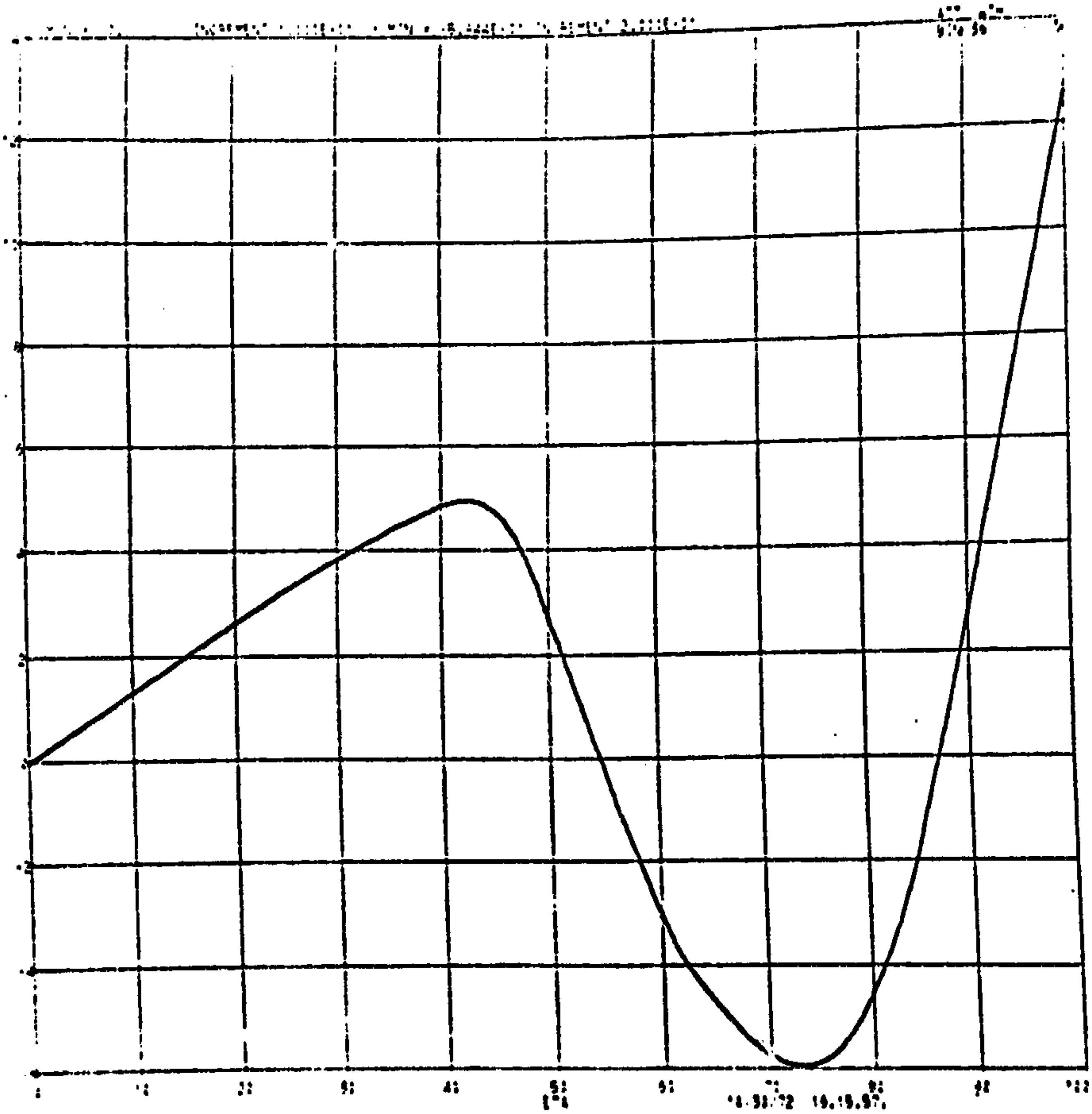


Figure II-9 Natural bending mode shape along wing elastic axis.
STOL wing (empty). ω = 21C.6 rad/sec.

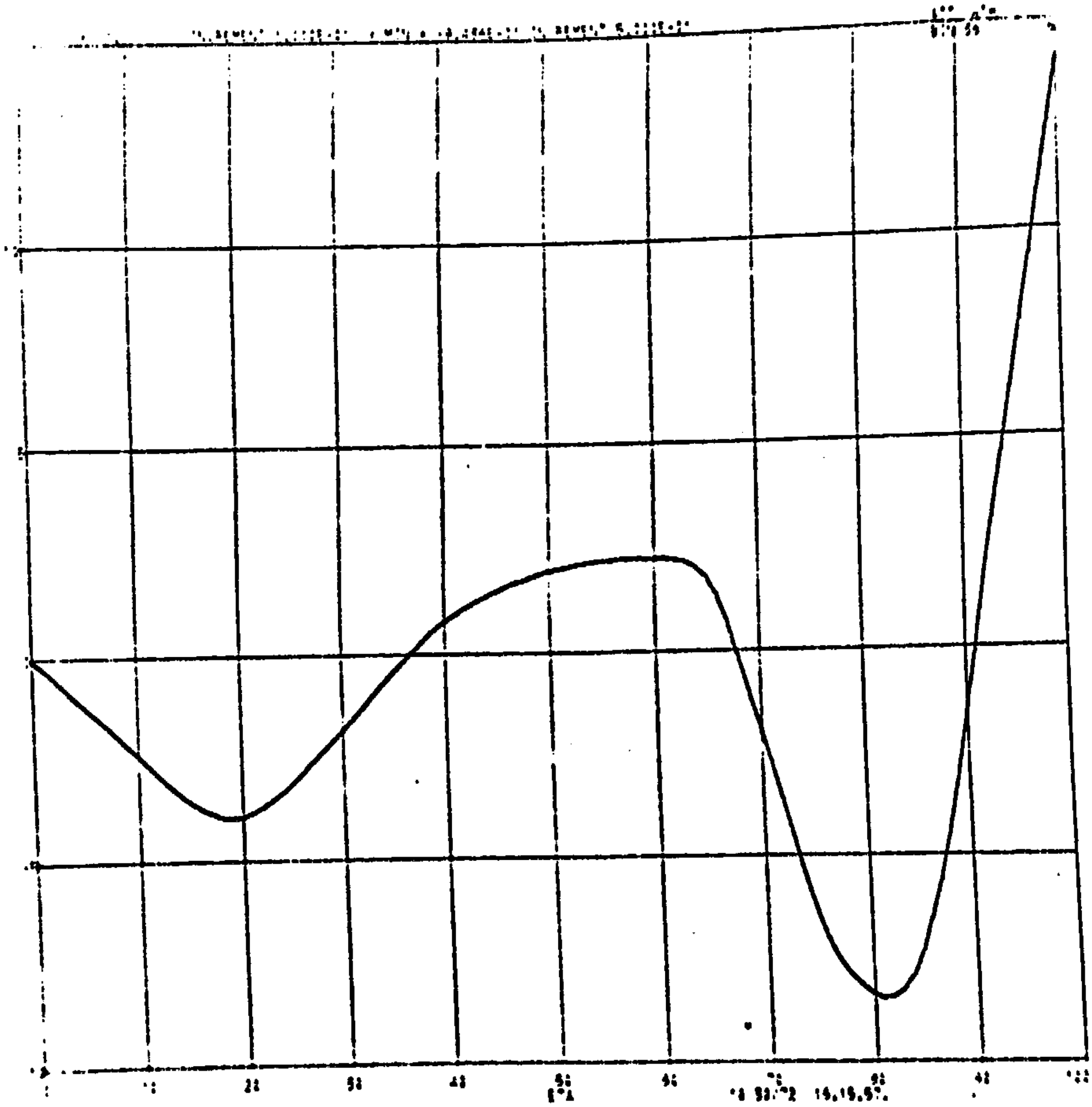


Figure II-10 Natural bending mode shape along wing elastic axis. STOL wing (empty). ω = 434.0 rad/sec.

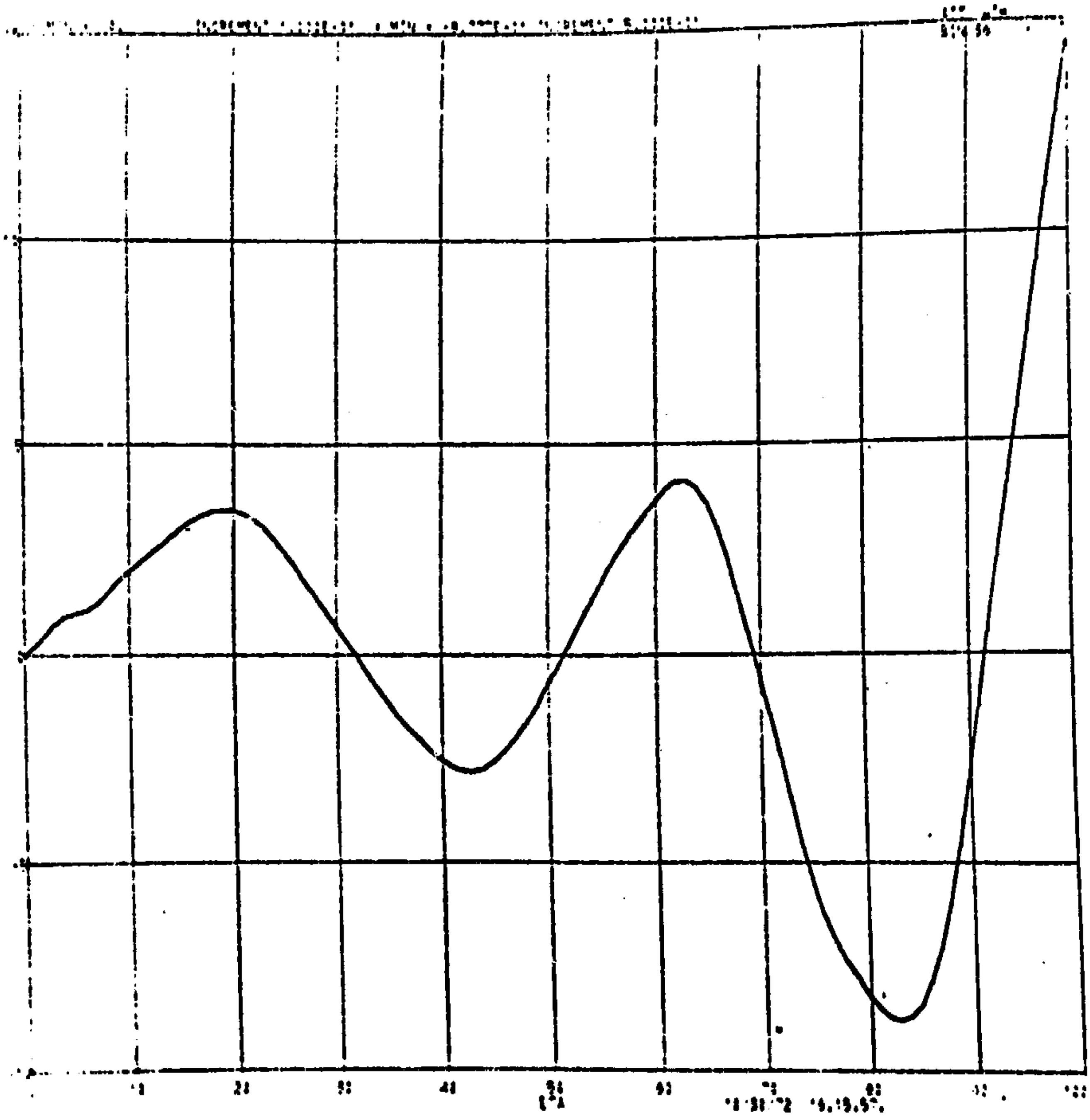


Figure II-11 Natural bending mode shape along wing elastic axis.
STOL wing (empty). ω = 513.1 rad/sec.

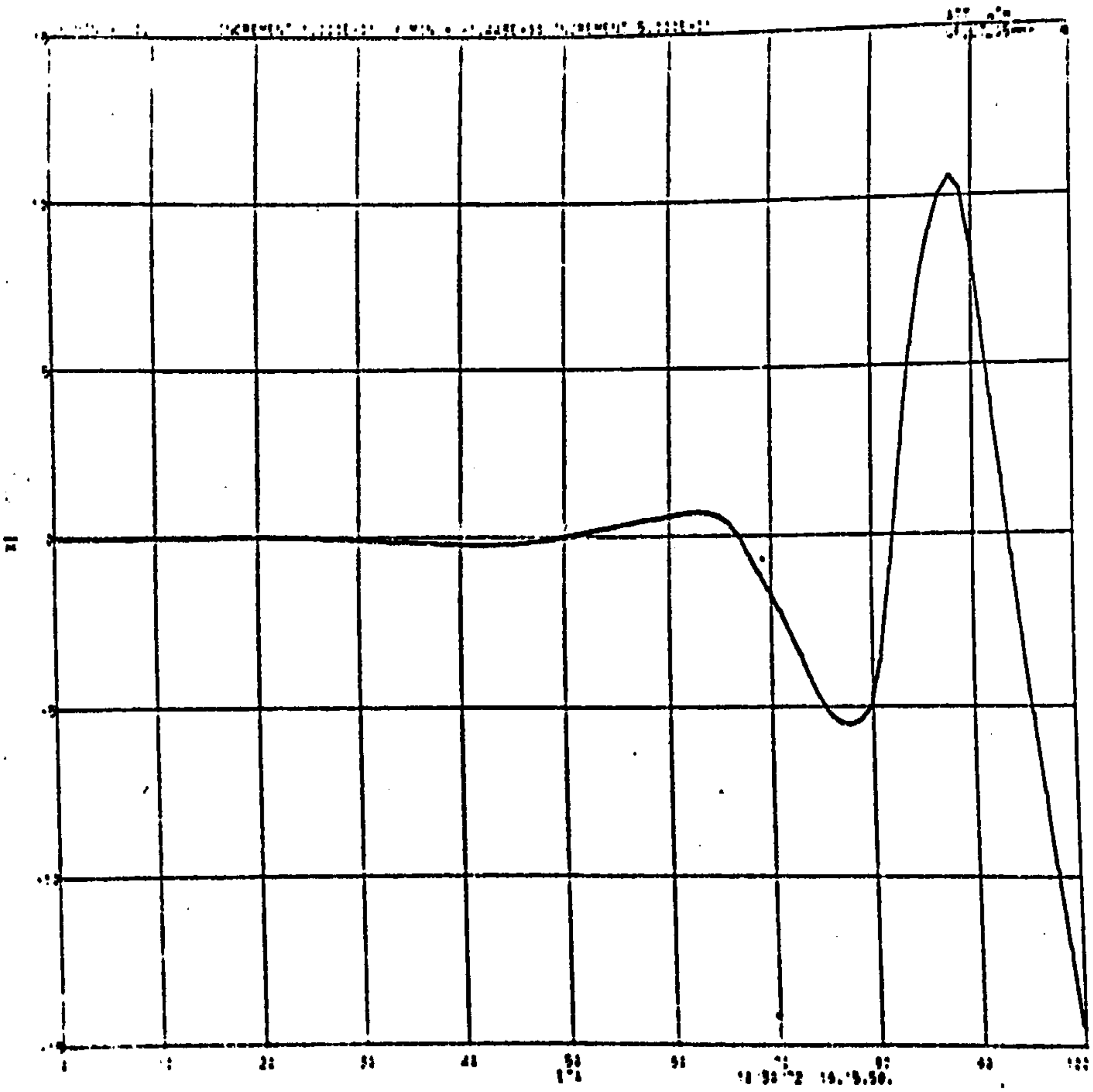


Figure II-12 Natural bending mode shape along wing elastic axis.
STOL wing (empty). ω = 1163 rad/sec.

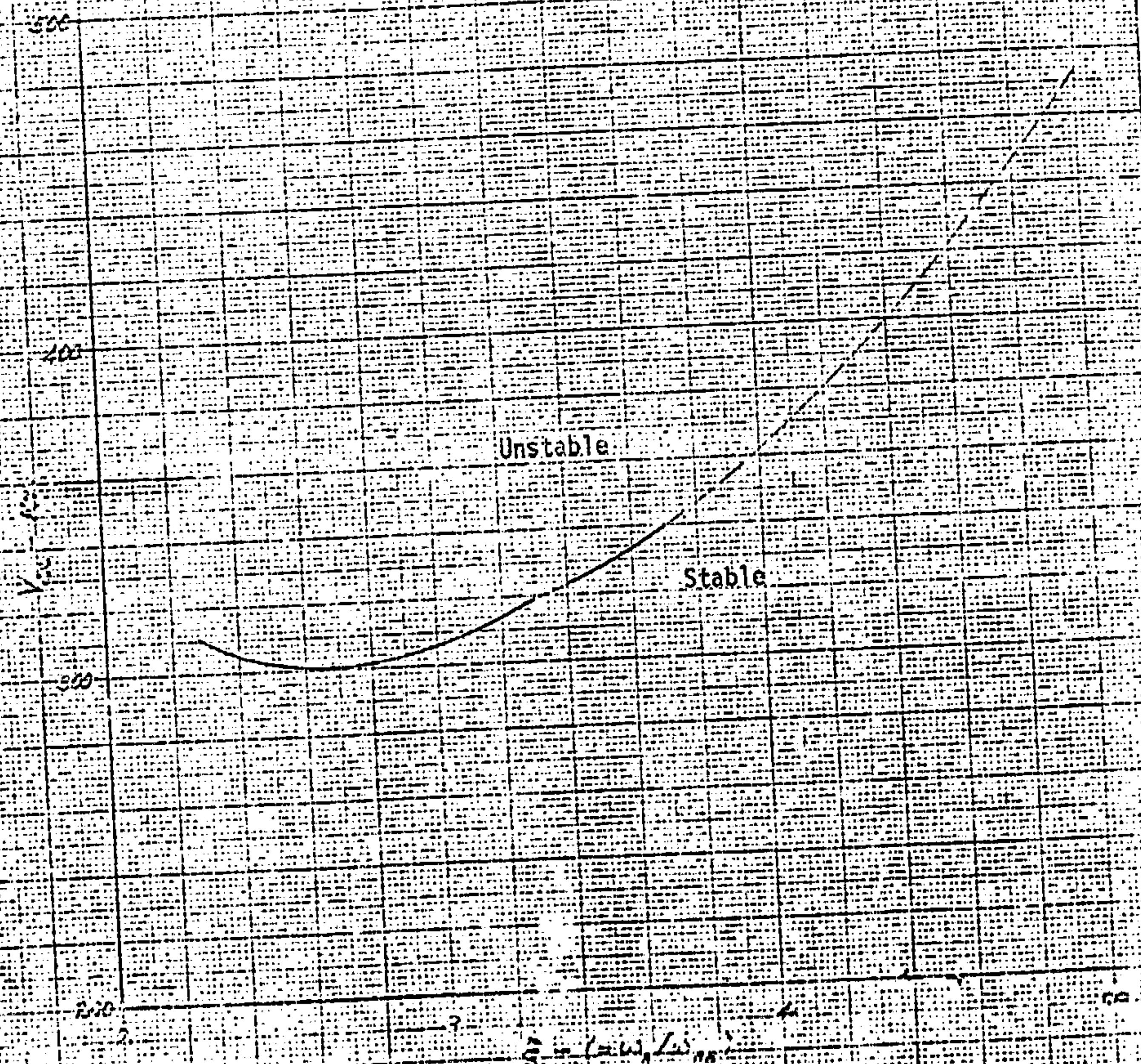


Figure II-13 Flutter boundary, STOL wing.
2015 lbi Nacelles.
ω_{WB} = 22.77 rad/sec.

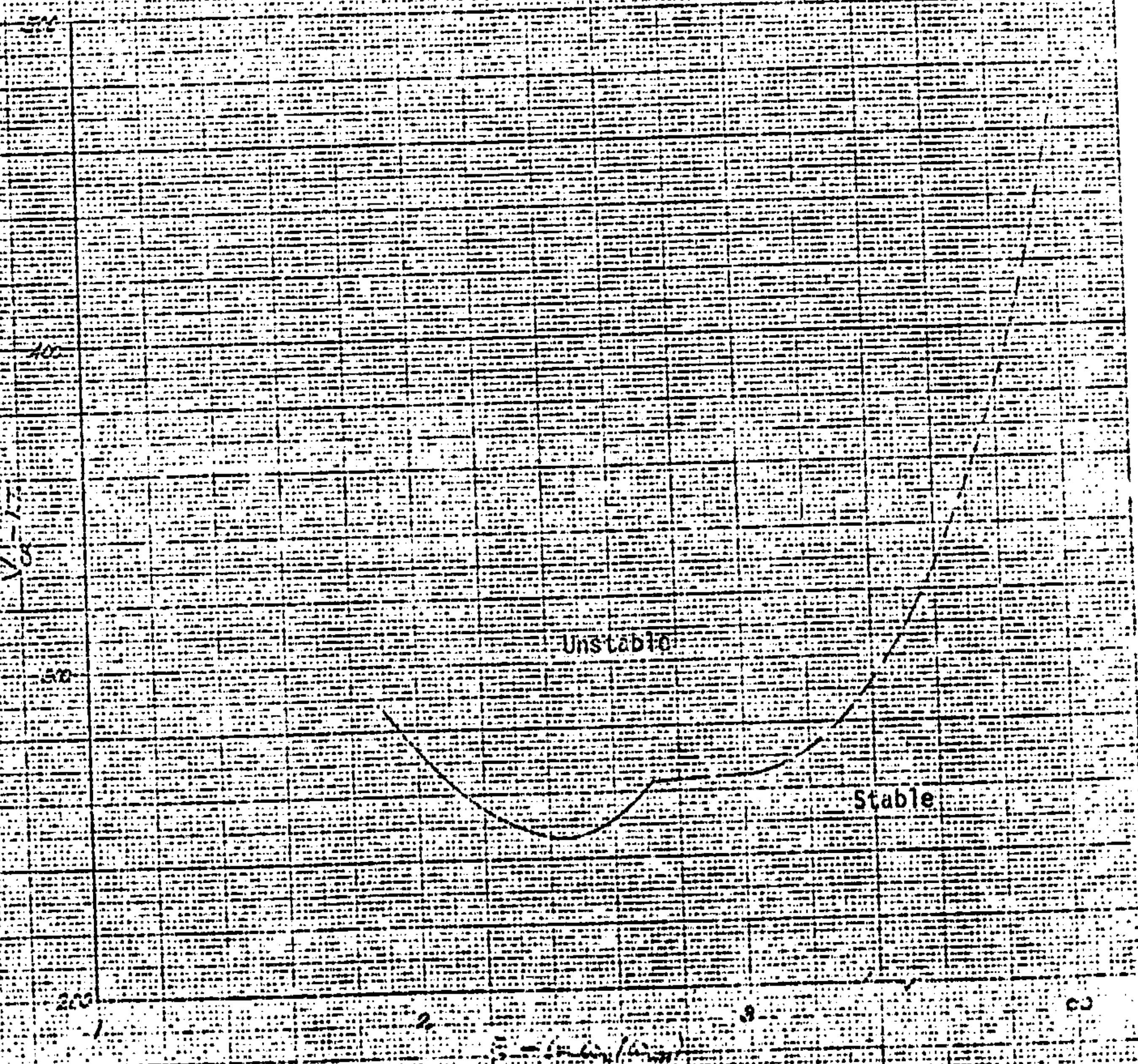


Figure II-14 Flutter Boundary, STOL Wing
350 lb. Nacelles
ω_{WB} = 21.57 rad/sec

Table II-1 Uncoupled Natural Frequencies - Wing with No Engines

Mode	Bending		Torsion	
	rad/sec	hz	rad/sec	hz
1	24.73	3.93	55.55	8.84
2	85.42	13.59	105.2	16.74
3	204.5	32.5	145.4	23.1
4	425.6	67.7	165.4	26.3

Table II-2 Uncoupled Natural Frequencies - Wing with 2015 lb. Nacelles, Attached Rigidly

Mode	Bending		Torsion	
	rad/sec	hz	rad/sec	hz
1	22.77	3.62	21.63	3.44
2	63.29	10.07	51.41	8.18
3	170.3	27.1	83.82	13.34
4	284.8	45.3	157.6	25.1

Table II-3 Uncoupled Natural Frequencies - Wing with 3500 lb. Nacelles, Attached Rigidly

Mode	Bending		Torsion	
	rad/sec	hz	rad/sec	hz
1	21.57	3.43	17.14	2.73
2	56.96	9.06	40.96	6.52
3	163.3	26.0	83.0	13.21
4	244.3	38.9	157.5	25.1

Table II-4 Bending Natural Frequencies - Wing with 2015 lb. Nacelles, Elastically Attached

Mode	rad/sec	hz
1	18.49	2.94
2	35.68	5.68
3	58.73	9.35
4	104.9	16.69
5	218.6	34.8
6	434.0	69.1
7	513.1	81.7
8	1163.3	185.1

Table II-5 Coupled Natural Frequencies - Wing with 2015 lb. Nacelles, Elastically Attached

Mode	rad/sec	hz
1	17.31	2.75
2	22.50	3.58
3	41.31	6.57
4	44.65	7.11
5	79.38	12.63
6	97.23	15.47
7	129.8	20.7
8	200.2	31.9
9	261.2	41.6
10	381.7	60.7
11	479.4	76.3
12	517.6	82.4
13	832.4	132.5
14	1969.0	313.4

Table II-6 STOL Wing Flutter Speeds - With 2015 lb. Nacelles, Elastically Mounted on the Wing. Pylon stiffness Reduced to 25% of Design Value.

Altitude ft.	Fuel	k	ω_F rad/sec	V_F ft/sec	V_F knots
0	empty	.25	35.5	613	363
10000		.23	35.3	662	392
20000		.20	35.2	759	449
25000		.19	35.1	795	471
30000	empty	.17	35.1	891	528
0	25%	.27	34.8	556	329
10000		.24	34.7	624	369
20000		.21	34.6	711	421
25000		.19	34.6	785	465
30000	25%	.18	34.6	828	490
0	50%	.28	34.4	529	313
10000		.25	34.3	592	351
20000		.22	34.3	671	397
25000		.20	34.2	737	436
30000	50%	.18	34.2	818	484
0	75%	.28	34.0	524	310
10000		.25	34.0	587	348
20000		.21	33.9	697	413
25000		.19	33.9	769	455
30000	75%	.18	33.9	812	481
0	100%	.25	33.8	582	345
10000		.22	33.8	661	391
20000		.18	33.7	807	478
25000		.17	33.7	855	506
30000	100%	.15	33.7	967	573

Table II-7 STOL Wing Flutter Speeds - With 3500 lb. Nacelles, Elastically Mounted on the Wing. Pylon stiffness Reduced to 25% of Design Value.

Altitude	Fuel	k	ω_F	V_F	
ft			rad/sec	ft/sec	knots
0	empty	.21	28.4	582	345
10000		.19	28.1	638	378
20000		.16	28.2	760	450
25000		.15	28.1	809	479
30000	empty	.14	28.1	864	512
0	25%	.22	27.9	546	323
10000		.19	27.8	631	374
20000		.17	27.7	702	416
25000		.16	27.6	743	440
30000	25%	.15	27.5	791	468
0	50%	.23	27.5	515	305
10000		.20	27.4	591	350
20000		.18	27.3	653	387
25000		.16	27.3	736	436
30000	50%	.15	27.3	784	464
0	75%	.23	27.2	509	301
10000		.21	27.1	557	330
20000		.18	27.1	648	384
25000		.17	27.0	685	405
30000	75%	.15	27.0	777	460
0	100%	.24	27.0	484	287
10000		.21	26.9	552	327
20000		.18	26.8	643	381
25000		.17	26.8	681	403
30000	100%	.16	26.8	723	428

Table II-8 STOL Wing Flutter Speeds - With 2015 lb. Nacelles, Elastically Mounted on the Wing. Pylon stiffness Reduced to 50% of Design Value.

Altitude ft	Fuel	k	ω_F rad/sec	V_F ft/sec	V_F knots
0	empty	.33	41.0	535	317
10000		.30	41.0	588	348
20000		.26	40.9	679	402
25000		.25	40.9	706	418
30000	empty	.23	40.9	767	454
0	25%	.34	40.6	514	304
10000		.30	40.5	582	345
20000		.26	40.5	672	398
25000		.24	40.5	727	430
30000	25%	.22	40.5	793	470
0	50%	.26	40.0	663	393
10000		.23	40.0	749	443
20000		.15	30.8	887	525
25000		.14	30.9	952	564
30000	50%	.12	29.4	1056	625
0	75%	.18	29.5	481	418
10000		.16	29.4	540	469
20000		.13	28.0	635	552
25000		.12	27.9	683	593
30000	75%	.11	27.6	738	641
0	100%	.18	29.4	703	416
10000		.15	28.4	816	483
20000		.13	28.2	936	554
25000		.12	28.1	1010	598
30000	100%	.11	27.9	1094	648

Table II-9 STOL Wing Flutter Speeds - With 3500 lb. Nacelles, Elastically Mounted on the Wing. Pylon stiffness Reduced to 50% of Design Value.

Altitude ft	Fuel	k	F rad/sec	V_F ft/sec	V_F knots
0	empty	.24	32.2	578	342
10000		.22	32.2	631	374
20000		.19	32.2	730	432
25000		.18	32.2	770	456
30000	empty	.17	32.2	815	483
0	25%	.29	32.0	476	282
10000		.26	32.0	531	314
20000		.23	32.0	600	355
25000		.22	32.0	627	371
30000	25%	.20	32.0	690	409
0	50%	.33	31.8	416	246
10000		.29	31.8	473	280
20000		.26	31.8	527	312
25000		.24	31.8	571	338
30000	50%	.22	31.8	623	369
0	75%	.30	31.6	310	269
10000		.27	31.6	344	299
20000		.23	31.6	404	351
25000		.21	31.6	442	384
30000	75%	.19	31.6	488	424
0	100%	.21	31.3	438	381
10000		.19	31.3	484	420
20000		.16	31.3	575	499
25000		.14	31.2	656	570
30000	100%	.13	31.2	706	613

Table II-10 STOL Wing Flutter Speeds - With 2015 lb. Nacelles, Elastically Mounted on the Wir . Pylon stiffness Reduced to 75% of Design Value.

Altitude ft	Fuel	k	ω_F rad/sec	V_F ft/sec	V_F knots
0	empty	.35	43.7	538	319
10000		.32	43.7	588	348
20000		.29	43.7	649	384
25000		.27	43.7	697	413
30000	empty	.25	43.6	752	445
0	25%	.30	43.1	619	367
10000		.27	43.0	687	407
20000		.23	43.0	806	477
25000		.22	43.0	843	499
30000	25%	.20	43.0	926	548
0	50%	.20	32.2	695	412
10000		.17	31.5	797	472
20000		.15	31.5	905	536
25000		.13	30.1	998	591
30000	50%	.12	29.9	1073	635
0	75%	.18	30.3	724	429
10000		.16	30.3	815	483
20000		.13	28.9	958	567
25000		.12	28.7	1031	610
30000	75%	.11	28.4	1115	660
0	100%	.18	29.9	716	424
10000		.15	28.9	831	492
20000		.13	28.8	954	565
25000		.12	28.7	1030	610
30000	100%	.10	26.8	1155	684

Table II-11 STOL Wing Flutter Speeds - With 3500 lb. Nacelles, Elastically Mounted on the Wing. Pylon stiffness Reduced to 75% of Design Value.

Altitude ft	Fuel	k	ω_F rad/sec	V_F ft/sec	V_F knots
0	empty	.25	34.3	592	351
10000		.23	34.3	643	381
20000		.21	34.3	705	417
25000		.20	34.3	740	438
30000	empty	.19	34.3	779	461
0	25%	.33	34.1	445	263
10000		.31	34.1	474	281
20000		.28	34.1	525	311
25000		.26	34.1	566	335
30000	25%	.24	34.1	612	362
0	50%	.17	33.8	470	278
10000		.15	33.8	521	308
20000		.13	33.8	583	345
25000		.12	33.8	634	375
30000	50%	.11	33.8	694	411
0	75%	.17	26.2	664	393
10000		.15	26.0	747	442
20000		.13	25.8	854	506
25000		.12	25.6	920	545
30000	75%	.11	25.4	995	589
0	100%	.17	25.8	654	387
10000		.14	24.9	766	454
20000		.12	24.5	880	521
25000		.11	24.3	951	563
30000	100%	.10	24.0	1033	612

Table II-12 STOL Wing Flutter Speeds - With 2015 lb. Nacelles, Elastically Mounted on the Wing.

Altitude ft	Fuel	k	ω_F rad/sec	V_F ft/sec	V_F knots
0	empty	.34	45.1	572	339
10000		.32	45.1	608	360
20000		.28	45.1	694	411
25000		.27	45.1	720	426
30000	empty	.25	45.1	778	461
0	25%	.28	44.3	683	404
10000		.25	44.3	764	452
20000		.22	44.3	868	514
25000		.20	44.3	954	565
30000	25%	.18	44.3	1058	626
0	50%	.20	32.6	479	416
10000		.17	31.7	549	477
20000		.14	30.5	642	557
25000		.13	30.4	687	596
30000	50%	.12	30.1	738	641
0	75%	.18	30.5	731	433
10000		.16	30.6	823	487
20000		.13	29.2	967	573
25000		.12	29.0	1041	616
30000	75%	.11	28.7	1126	667
0	100%	.17	42.0	752	445
10000		.15	29.6	849	503
20000		.13	29.4	976	578
25000		.11	27.6	1084	642
30000	100%	.10	27.3	1176	696

Table II-13 STOL Wing Flutter Speeds - With 3500 lb. Nacelles, Elastically Mounted on the Wing.

Altitude ft	Fuel	k	ω_F rad/sec	V_F ft/sec	V_F knots
0	empty	.25	35.5	612	362
10000		.24	35.5	638	378
20000		.23	35.5	666	394
25000		.22	35.5	696	412
30000	empty	.20	35.5	766	454
0	25%	.34	35.3	447	265
10000		.32	35.3	475	281
20000		.29	35.3	524	310
25000		.27	35.2	563	333
30000	25%	.25	35.2	608	360
0	50%	.27	34.9	557	330
10000		.24	34.9	626	371
20000		.22	34.9	683	404
25000		.20	34.8	751	445
30000	50%	.18	34.8	834	494
0	75%	.17	26.4	670	397
10000		.15	26.3	754	446
20000		.13	26.1	864	512
25000		.11	25.1	984	583
30000	75%	.10	24.9	1074	636
0	100%	.16	25.4	684	405
10000		.14	25.1	773	458
20000		.12	24.7	889	526
25000		.11	24.5	959	568
30000	100%	.10	24.2	1042	617

Section III. STRESS

This section contains the preliminary analyses of the wing box, flaps and engine pylon structure. Included are the stiffnesses and approximate section properties required for the vibration and flutter analyses.

The wing box analysis begins on page 44. Bending, torsional and vertical shear stiffnesses are on pages 46 and 49. An internal bending load distribution in the box covers is on page 50.

The flap analysis begins on page 44. Pages 58 and 59 summarize the flap bending and torsional stiffnesses. Analysis and section properties of the flap support structure are provided on pages 73 through 100. The figure on page 100 identifies the typical sections for the inboard, center and outboard flap supports.

The pylon analysis begins on page 103. Vertical shear stiffness, bending stiffness and torsional stiffness are plotted versus pylon length on pages 106, 107, and 108, respectively.

Wing Box Structure

The wing box is a single cell box beam of skin-stringer construction. Material is aluminum alloy (2024-T3 and 7075-T6). Method of analysis is at a preliminary design level assuming an allowable bending tensile stress in the lower surface of 50,000 psi. Upper surface wing bending area is based on the lower surface area arbitrarily increased 20% to account for a normally higher area due to the upper surface being buckling critical. The allowable stress of 50,000 psi accounts for area out due to holes, combining bending stresses with shear stresses and fatigue considerations.

Resultant bending material determined the bending stiffness. The vertical shear stiffness is based on the total estimated spar web thicknesses. Torsional stiffness is based on the spar web thicknesses and the upper and lower skin thicknesses. Skin thicknesses were determined by assuming 50% of the bending material to be shear material.

Flap Loads

The normal force (F_N) and pitching moments on flap numbers 1, 2, and 3 for inboard, center and outboard sections were determined from the coefficients shown on pages 101 and 102. These forces and moments are reacted at the flap supports. Bending moments on the flaps are used to determine the skin gages. The bending and torsional rigidity curves for the flaps are shown on pages 58 and 59, respectively. The calculations for the forces and moments are shown on pages 61 through 72, and are summarized on page 60.

The reactive loads and moments are applied to the flap support beams and tracks to determine the size of the cross sections. Bearing sizes capable of carrying the required loads influence the initial sizing of the members. The bearing loads applied to the flanges of the beams and tracks are critical. The shape and sizes of several

critical cross sections are shown for each section of the flaps. In all cases the dimensions, cross sectional areas, moments of inertia of the beams and tracks are varying. In the absence of detailed drawings, it should be sufficiently accurate to consider a linear variation of properties from section to section.

Flap Structure

The material of construction for the skin and frames of the flaps is 17-7 PH stainless steel. The skins are brazed to a stainless steel honeycomb core. The tracks, supports, and fittings are made from 17-4 PH stainless steel. The mechanical properties for these materials are referenced in MIL-HDBK-5, and they are reproduced for convenience below.

17-7 PH stainless steel*

MIL-S-25043

Sheet

F_{tu} = 177 KSI

F_{ty} = 150 KSI

F_{cy} = 158 KSI

F_{su} = 115 KSI

E = 29.0 x 10^6 PSI

E_c = 30.0 x 10^6 PSI

17-4 PH Stainless Steel*

AMS 5643

Bar and Forging

F_{tu} = 190 KSI

F_{ty} = 170 KSI

F_{cy} = 178 KSI

F_{su} = 123 KSI

E = 29.0 x 10^6 PSI

E_c = 30.0 x 10^6 PSI

*Room temperature values

725 FT^2 STOL WING BENDING & TORSIONAL STIFFNESS

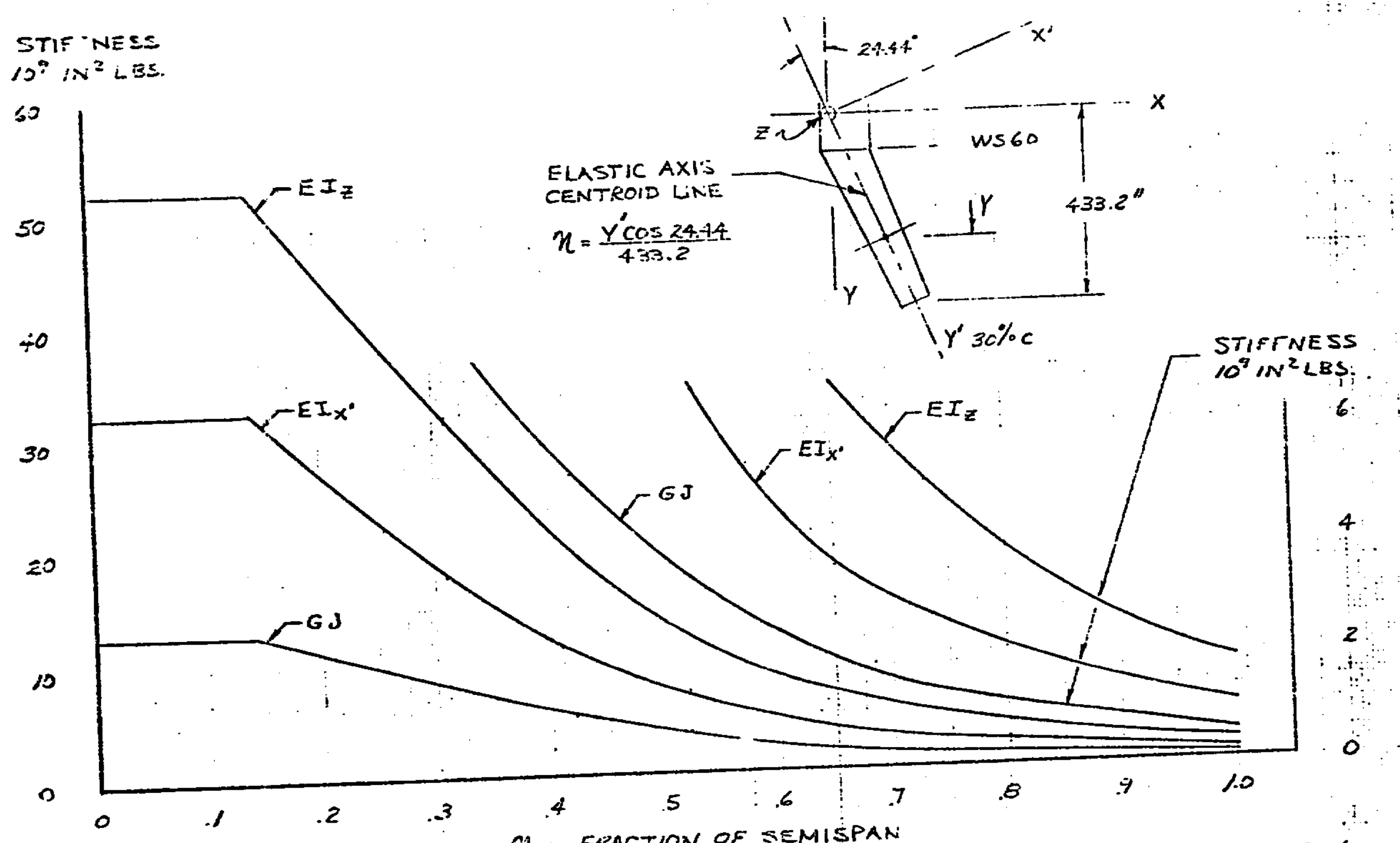


FIG. III-1

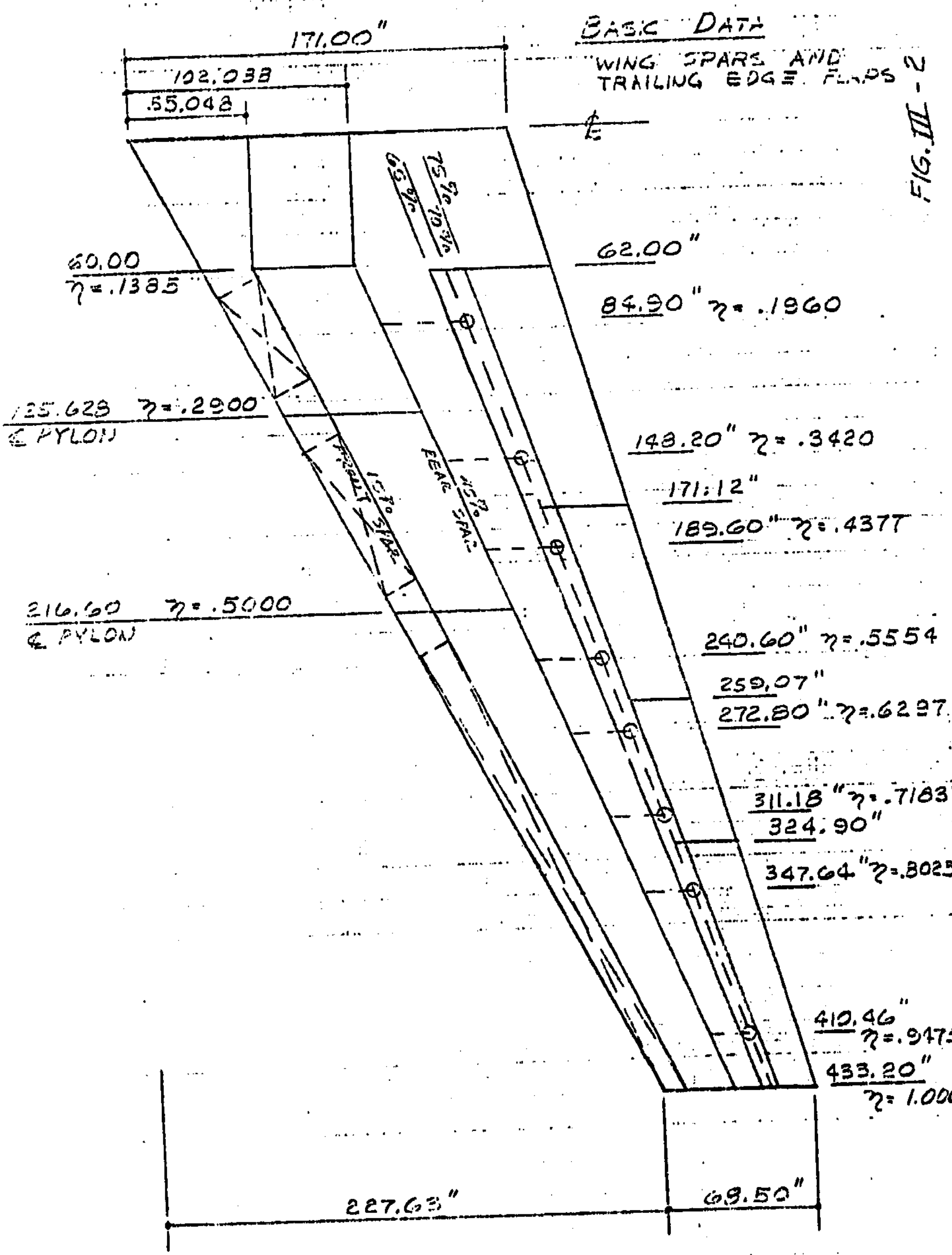
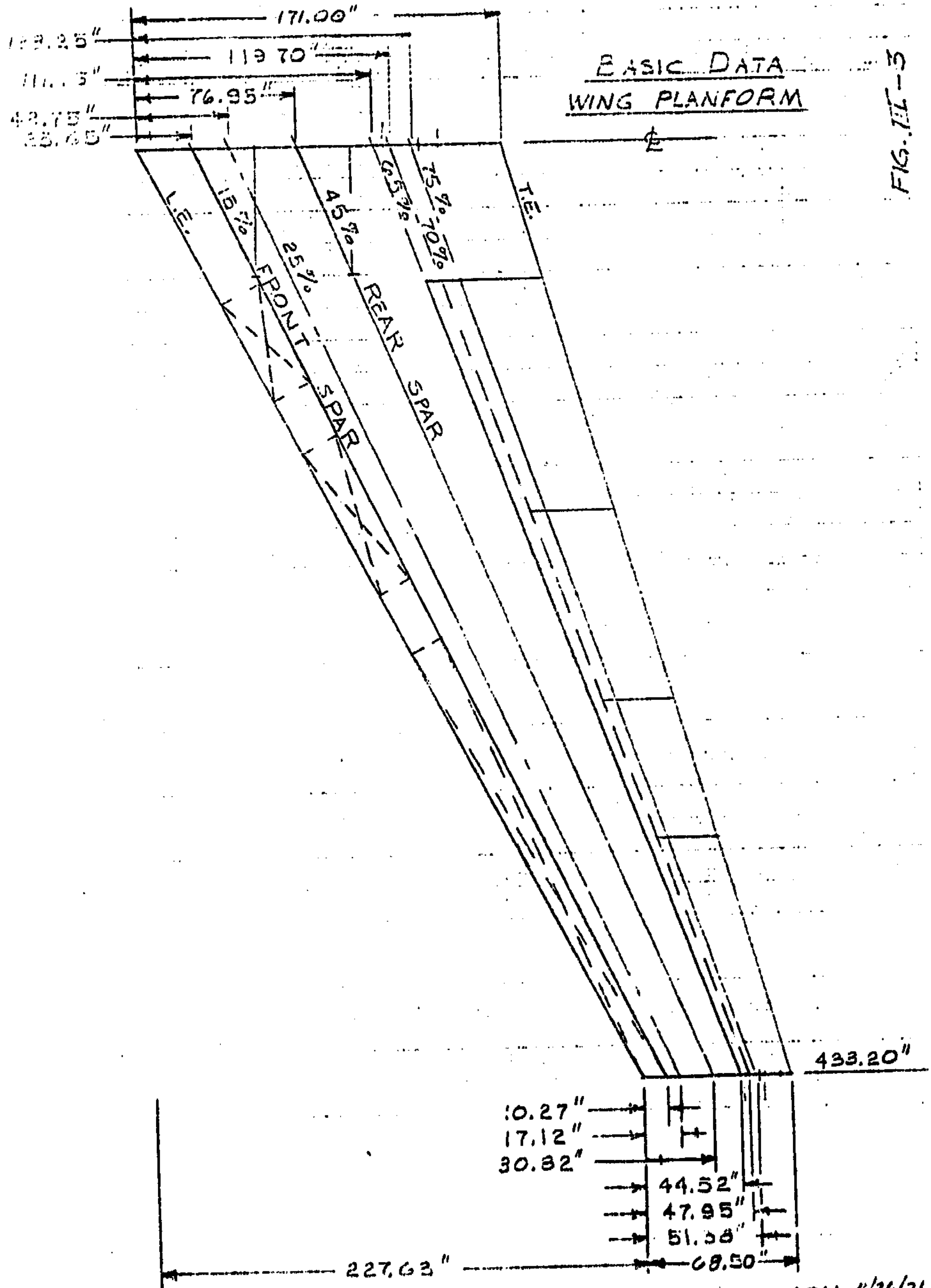


FIG. III-2

WTW 11/24/71



BASIC DATA
WING PLANFORM
℄
171.00"
119 70"
76.95"
42.75"
25.65"
L.E.
15%
25%
45%
65%
75%
70%
T.E.
FRONT SPAR
REAR SPAR
433.20"
10.27"
17.12"
30.82"
44.52"
47.95"
68.50"
227.63"
WTW 11/26/71

FIG. III-3

725 FT² STOL WING BOX

VERTICAL SHEAR STIFFNESS

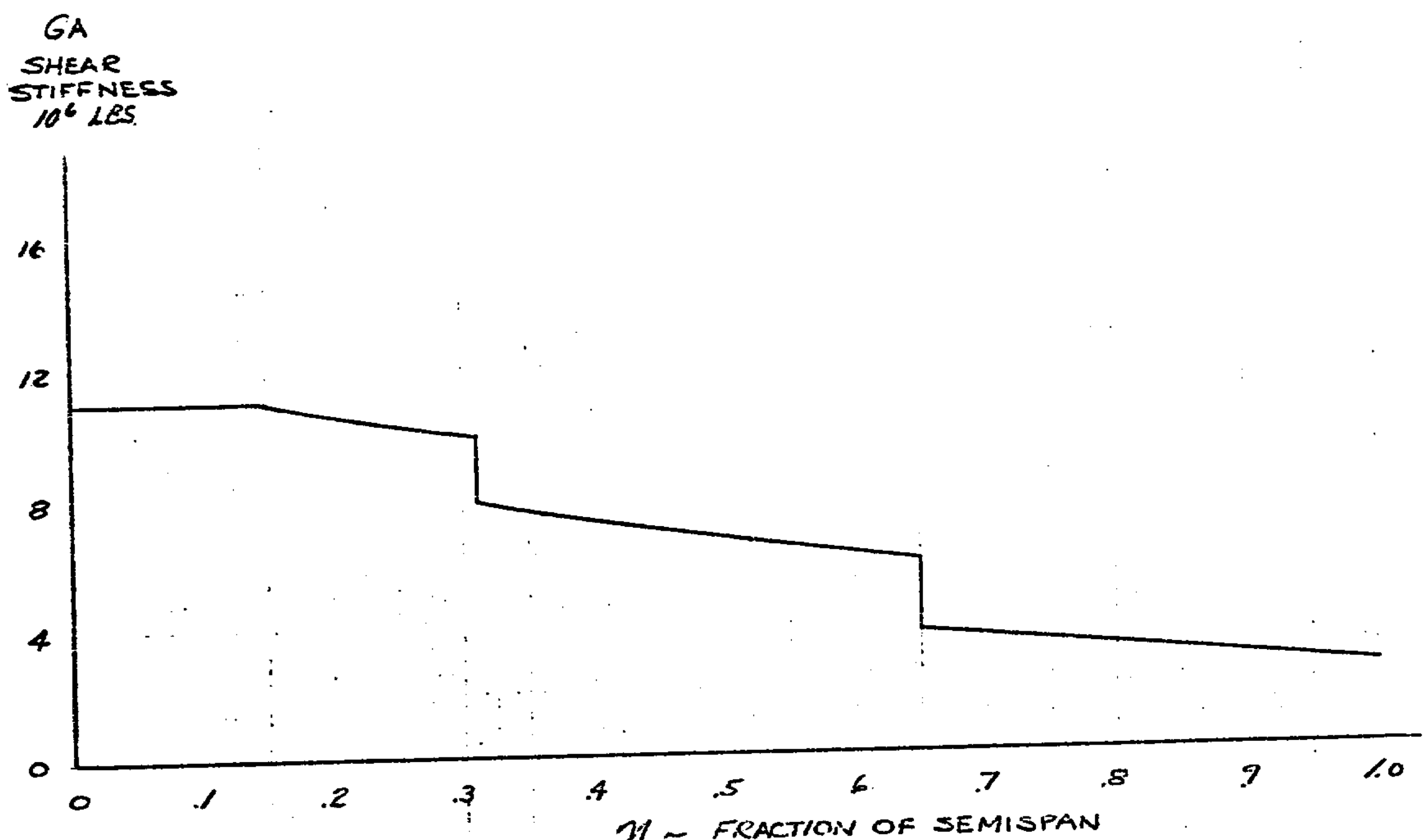


FIG. III-4

725 FT² STOL WING
INTERNAL BENDING LOADS (ULT.)

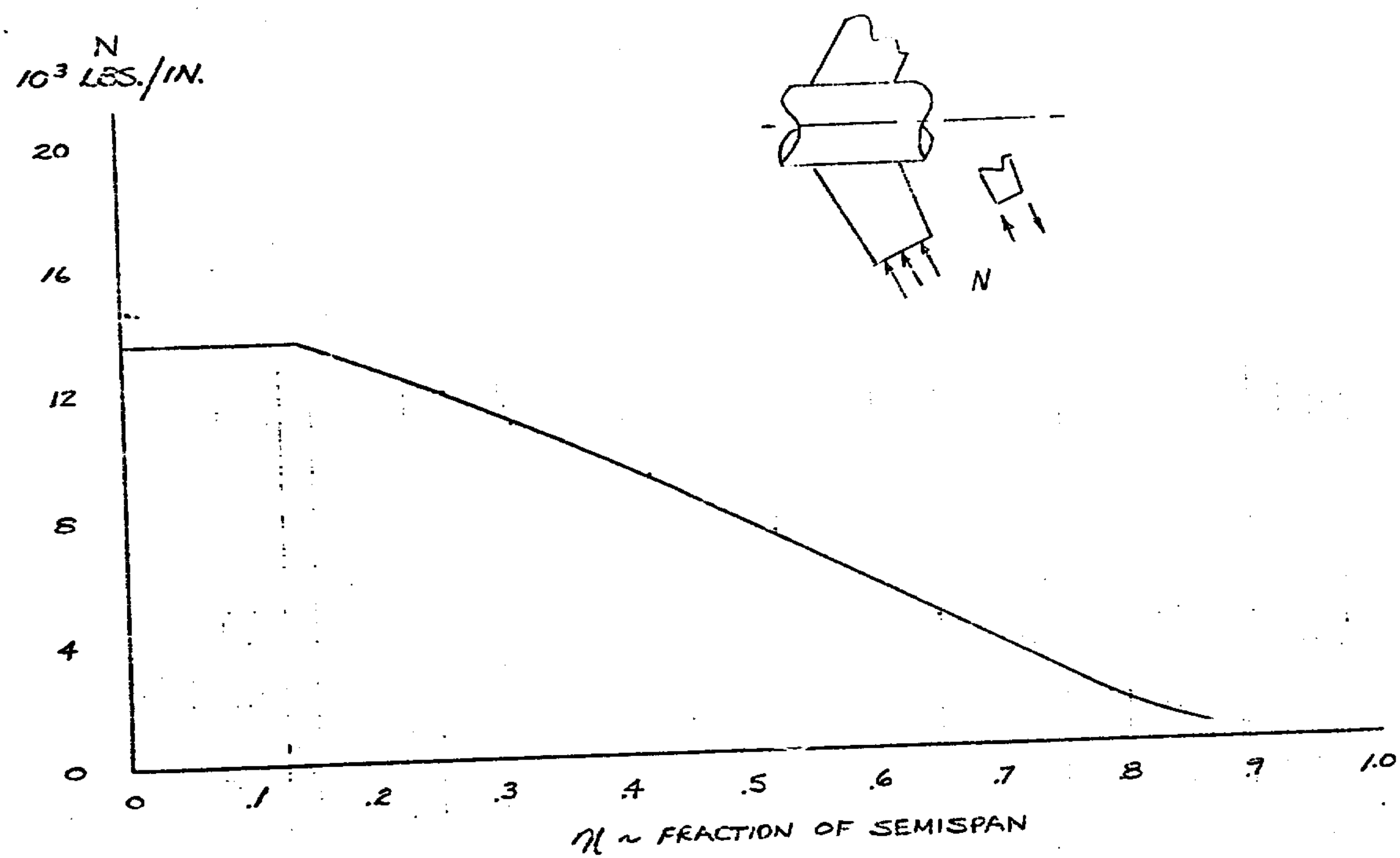


FIG III - 5

TABLE III-1

STOL WING BOX

η	N.S. TO 30% C	h_{avg} IN.	h_e IN.	B IN.	A_e IN^2	M_{ULT} 10^6 IN-LB.	V_{ULT} LBS.		
.123	58.53	25.2	22.90	43.1	1087	13.58	69,000		
.256	121.82	23.1	20.75	39.8	920	9.75	54,800		
.310	147.51	22.2	20.20	38.5	855	8.32	57.000		
.420	199.85	20.4	18.13	35.3	720	5.77	44,500		
.520	247.44	18.8	17.00	32.2	605	3.90	40,900		
.650	309.30	16.7	15.08	28.8	481	1.83	27,000		
.800	380.67	14.3	12.90	24.4	349	0.45	12,000		
1.000	475.84	11.1	9.90	18.8	209	0.00	0		

η = FRACTION OF SEMISPAN

N.S. = NORMAL STATION TO 30% CHORD LINE (INCHES) = $433.2\,\eta / \cos 24.44$

h_{avg} = AVERAGE BOX HEIGHT

h_e = EFFECTIVE BOX HEIGHT FOR BENDING = $.85 \times h_{MAX.}$

B = BOX WIDTH

A_e = ENCLOSED BOX AREA = $B \times h_{avg}$

M_{ULT} = ULTIMATE BENDING MOMENT.

V_{ULT} = ULTIMATE VERTICAL SHEAR.

TABLE III-2

STOL WING BOX

η	M_{ULT} 10^6 IN-LB	P_{TOT} LBS	N LBS/IN	A_L IN^2	A_U IN^2	V_{ULT} LBS	h_e (IN) EFF. BOX HGHT	q LBS/IN	t_w IN.
.122	12.58	593013	13,759	11.86	14.23	69,000	22.90	1507	.063
.256	9.75	469,880	11,806	9.40	11.23	54,800	20.75	1330	.063
.310	8.32	411,881	10,698	7.74	9.29	57,000	20.20	1411	.063 .050
.420	5.77	318,257	9,016	6.36	7.64	44,500	18.13	1227	.050
.520	3.90	229412	7125	4.59	5.51	40,900	17.00	1203	.050
.650	1.83	121,353	4214	2.43(2.90)	2.91(3.48)	27,000	15.08	895	.050 .032
.800	0.45	34,884	1430	.70(2.44)	.83(2.93)	12,000	12.90	465	.032
1.000	0.00	0	0	0 (1.98)	0 (2.26)	0	9.90	0	.032

MIN. GAUGE = .108 (t_{MIN} = .10 LOWER SURFACE)

M_{ULT} = ULT. BENDING MOMENT

P_{TOT} = SPANWISE LOAD IN COVER MATERIAL - LBS = M_{ULT}/h_e

N = " " " " " - LBS/IN = P_{TOT}/B

A_L = TOTAL LOWER COVER MATERIAL BASED ON 50,000 PSI TENSION ALLOWABLE

A_U = TOTAL UPPER COVER MATERIAL BASED ON ASSUMPTION OF 20% ADDITIONAL MATERIAL NEEDED FOR COMPRESSION

NOTE THAT THE 50,000 PSI ALLOWABLES ARE "FIRST CUT" SINCE THE FINAL ALLOWABLES CANNOT BE DETERMINED UNTIL AFTER SEVERAL ITERATIONS AND ESTABLISHMENT OF FINAL DETAIL DESIGN.

V_{ULT} = ULT. VERTICAL SHEAR

$q = V_{ULT}/2h_e$ = WEB SHEAR FLOW IN SPARS — ASSUME FRONT AND REAR SHEAR EQUAL

t_w = SPAR WEB THICKNESS SEE BUCKLING GRAPH

NASA-Langley Form 10 (AUG 1960)

TABLE [illegible]

STOL WING BOX

n	B	$\bar{t}_L$	$\bar{t}_U$	$\frac{(\bar{t}_L+\bar{t}_U)B^3}{12}$	$=\frac{(A_L+A_U)B^2}{12}$	A_w	$\frac{A_w B^2}{2}$	EI_z 10^9 IN² LBS	GA
.123	43.1	.275	.330	4039		1.44	1337	5376	11.2
.256	39.8	.236	.283	2730		1.31	1032	3769	10.2
.310	38.5	.214	.257	2239		1.27 / 1.01	941 / 743	3130 / 2477	7.9 / 7.9
.420	35.3	.180	.216	1454		.91	567	2021	7.1
.520	32.2	.143	.171	873		.85	441	1314	6.6
.650	28.8	.100	.120	441		.75 / .48	311 / 199	752 / [illegible]	5.9 / 3.7
.800	24.4	.100	.120	266		.41	122	388	3.2
1.000	18.8	.100	.120	122		.32	57	179	2.5

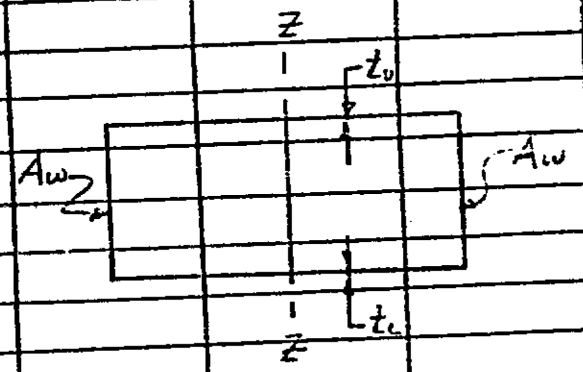
$\bar{t}_U$ = "SMEARED" COVER THICKNESS (IN.)

$\bar{t}_L$ = " " " "

A_w = SPAR WEB AREA (IN²) $h_e \times t_w$

$EI_z = \left[(\bar{t}_U+\bar{t}_L)B^3/12 + A_w B^2/2\right] 10^9$

$GA = 2 G A_w = 7.8 \times 10^6 \times A_w$ (FORM FACTOR = 1)



[illegible] 10 (AUG 1960)

STOL WING BOX

TABLE [illegible]

η	$A_U + A_L$ IN^2	A_U IN^2	h_e IN.	y_L IN.	y_U IN.	$A_U y_U^2$ IN^4	$A_L y_L^2$ IN^4	I_x' IN^4	EI_x' 10^9 IN^2-LB
.123	26.09	14.23	22.90	12.49	10.41	1542	1850	3392	33.92
.251	20.68	11.23	20.75	11.32	9.43	1003	1205		22.06
.310	18.13	9.89	19.91	10.86	9.05	810	972		17.22
.420	14.00	7.64	18.1[illegible]	9.89	8.24	519	622		11.41
.520	10.10	5.51	17.00	9.27	7.73	329	394		7.23
.650	6.38	3.48	15.08	8.23	6.85	163	196		3.59
.800	5.37	2.93	12.99	7.04	5.86	101	121		2.22
1.000	4.14	2.26	9.90	5.40	4.50	46	55		1.61

A_U = TOTAL UPPER COVER AREA

A_L = TOTAL LOWER COVER AREA

h_e = EFFECTIVE BOX HEIGHT = 85% OF MAX. BOX HEIGHT

y_L = DISTANCE FROM BOX CENTROID TO COVER CENTROID = $A_U \times h_e / A_U + A_L$

y_U = " " " " " " "

I_x' = $A_U y_U^2 + A_L y_L^2$ (MOMENT OF INERTIA)

E = 10.0×10^6 PSI MODULUS OF ELASTICITY ASSUMED FOR ALUM.

[illegible] 10 (AUG 1959)

STOL WING BOX

TABLE [illegible]

η	t_{SU} IN.	t_{SL} IN.	B/t_{SU}	B/t_{SL}	t_w IN.	$\frac{2h_{[illegible]}}{t_w}$	$\Sigma S/t$	$\frac{4GA_0^2}{10^9}$	[illegible]
.123	.165	.137	261	314	.063	800	1375	18432	13.41
.256	.141	.113	282	338	.063	735	1355	13204	9.74
.310	.128	.107	301	360	.063 / .050	705 / [illegible]	1366 / 1551	11404	8.34 / [illegible]
.420	.108	.090	327	392	.050	818	1537	8087	5.26
.520	.085	.071	379	453	.050	752	1584	5710	3.60
.650	.060	.050	480	576	.050 / .032	628 / 1042	1724 / 2098	3609	2.09 / 1.72
.800	.060	.050	407	488	.032	894	1789	1900	1.06
1.000	.060	.050	313	376	.032	594	1383	681	.47

$t_{SU} = A_U/2B$ ASSUMED 50% COVER AREA IN SKIN — BALANCE IN STRINGERS & CAPS

$t_{SL} = A_L/2B$ " " " " " " " " " "

t_w = SPAR WEB THICKNESSES (ASSUMED EQUAL)

$G = 3.9 \times 10^6$ PSI MODULUS OF RIGIDITY FOR ALUM.

STOL

WING SHEAR

(RIGID)

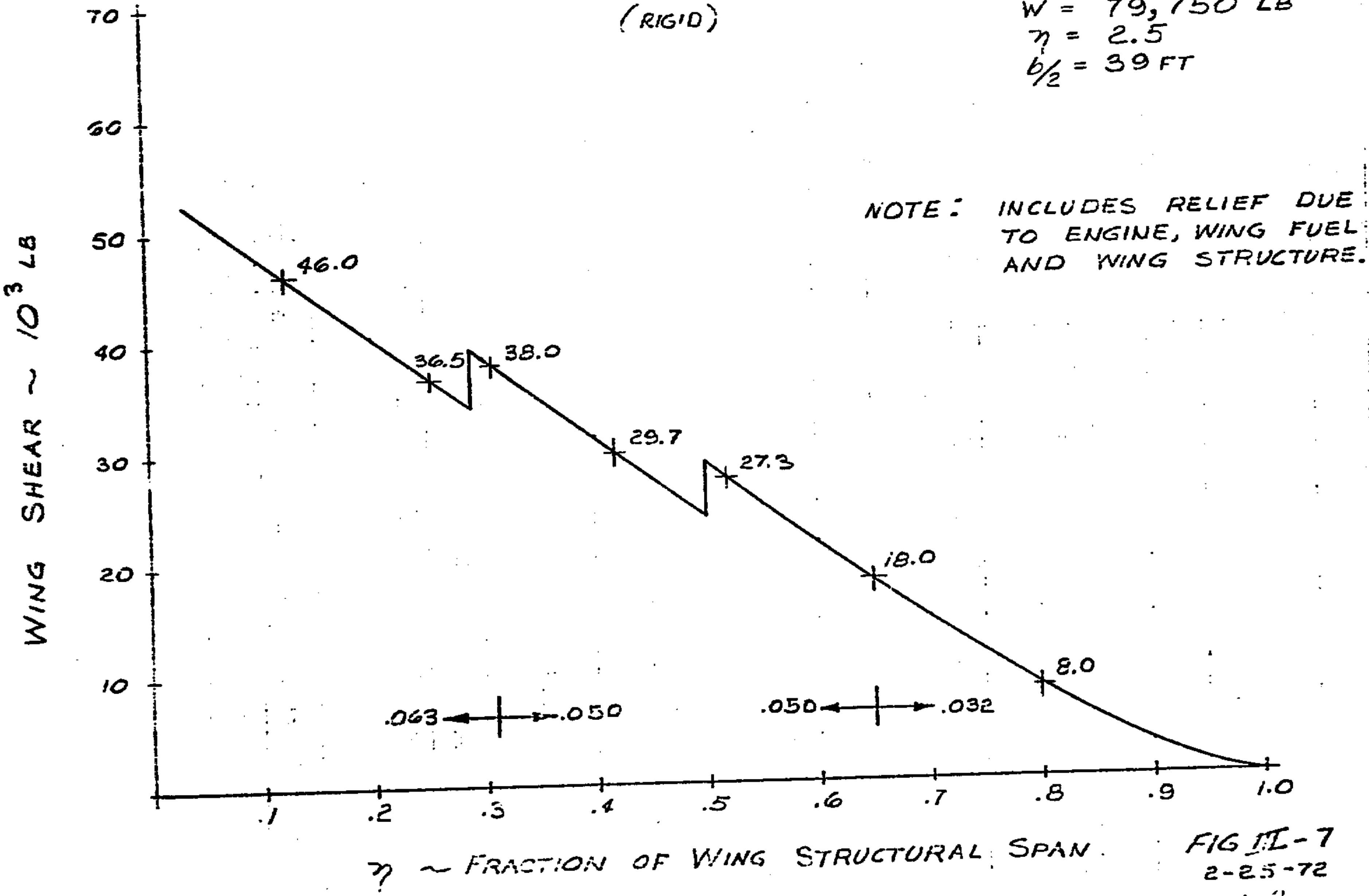


FIG III-7
2-25-72

STOL

WING BENDING MOMENT
(RIGID)

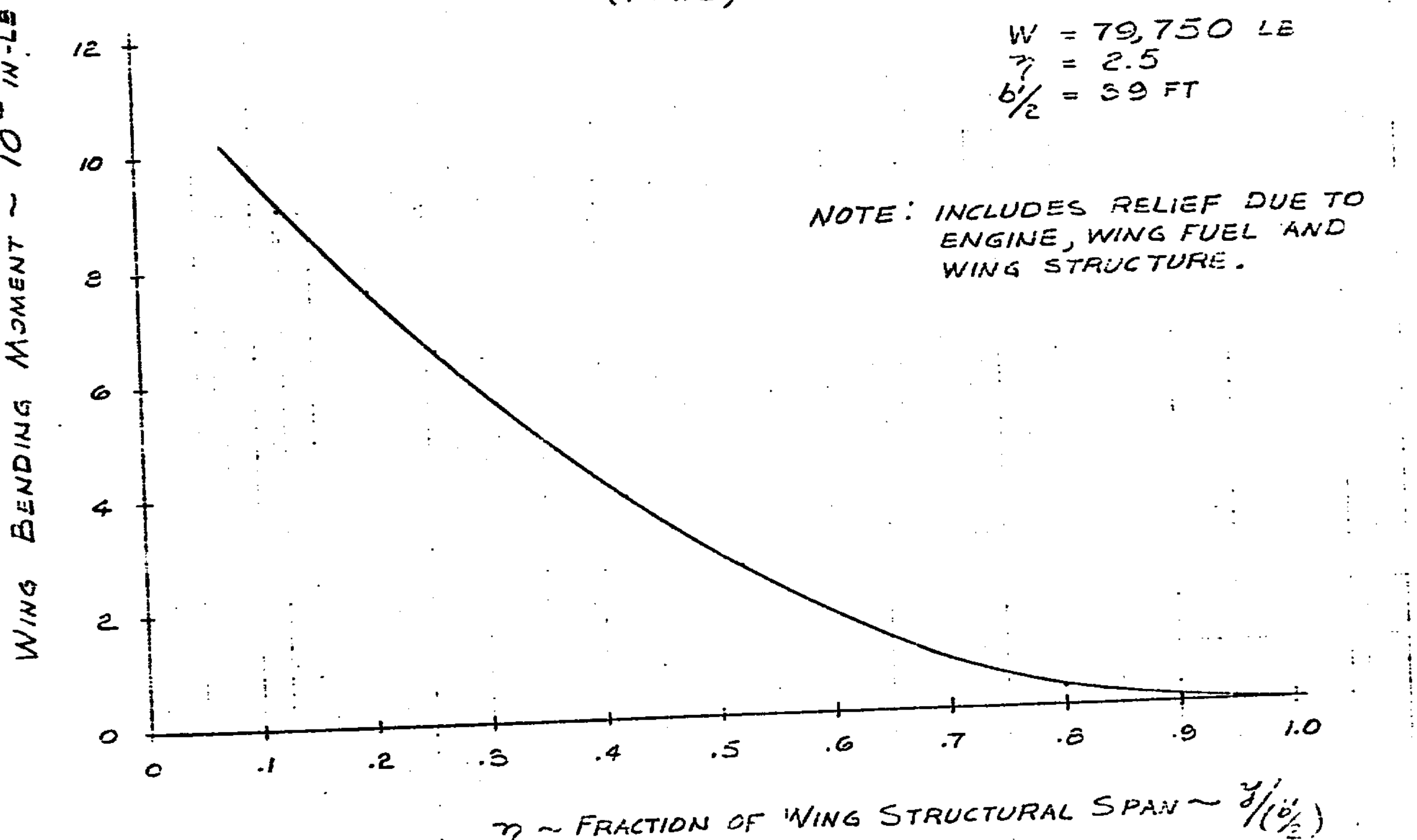
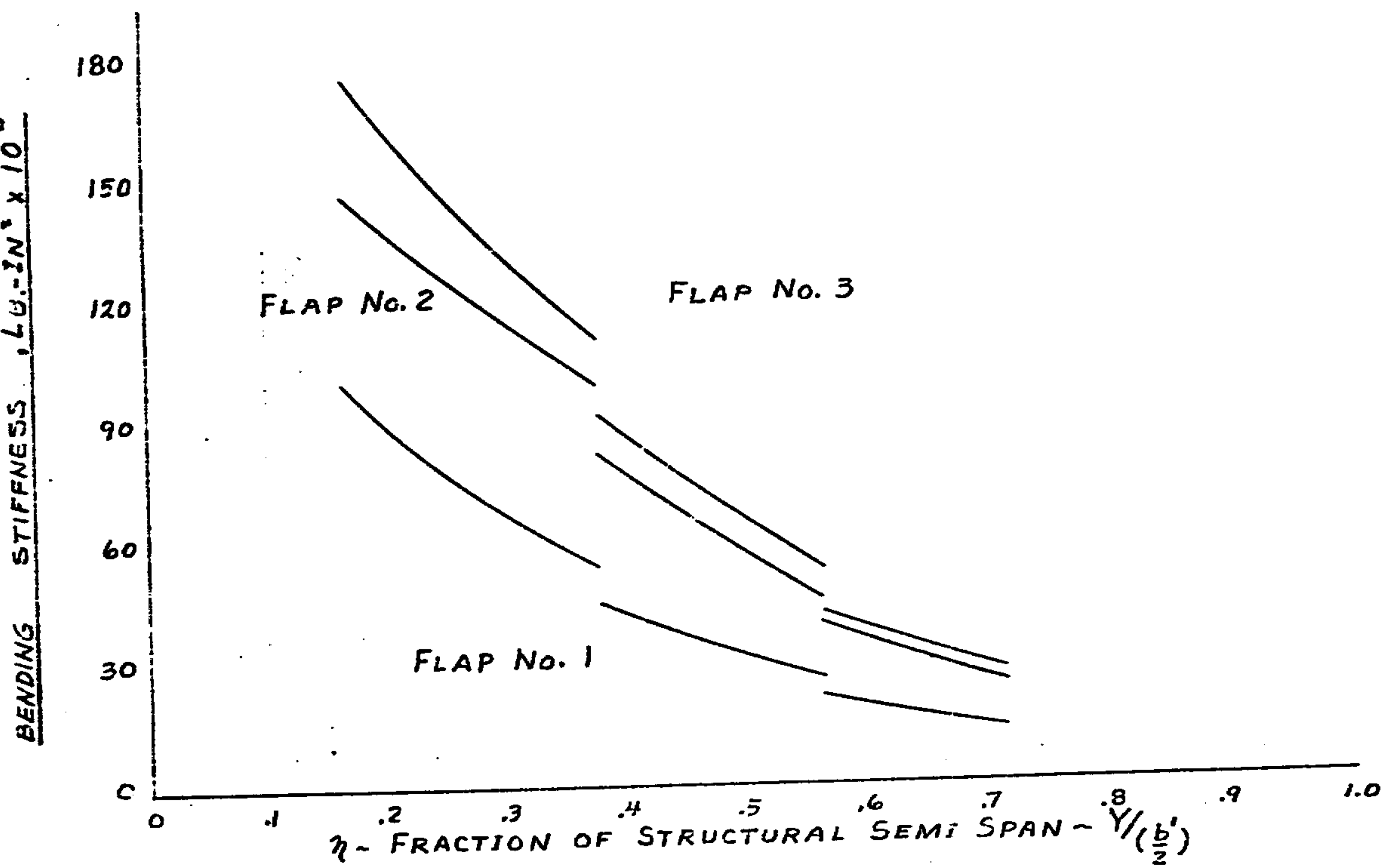


FIG. III-8

STOL



BENDING STIFFNESS , LB.-IN² x 10⁶
180
150
120
90
60
30
0
FLAP No. 2
FLAP No. 3
FLAP No. 1
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
η ~ FRACTION OF STRUCTURAL SEMI SPAN ~ Y/(b'/2)

FIG II - 9

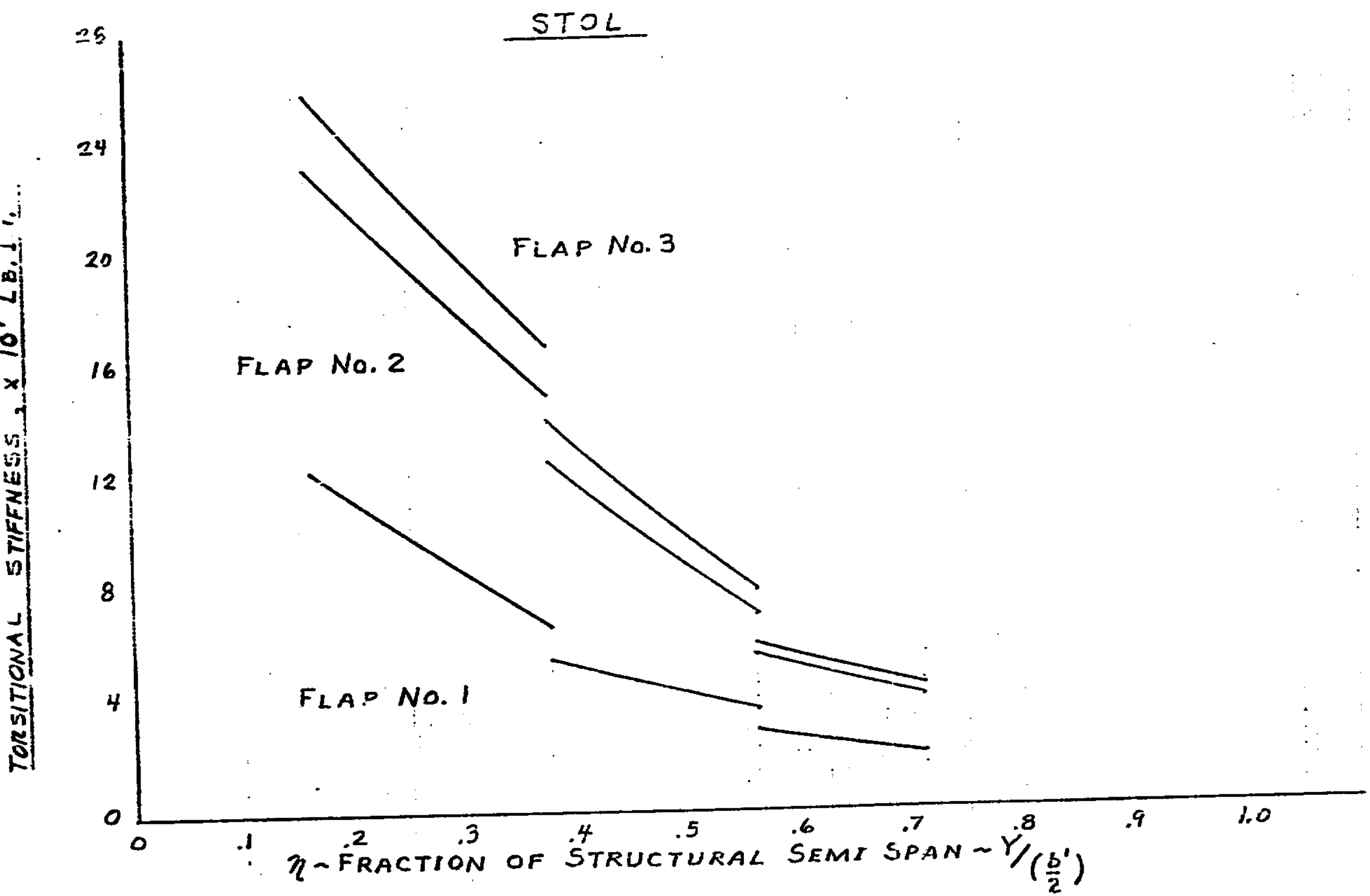
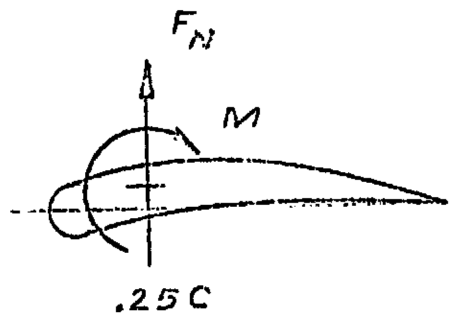

STOL
FLAP No. 3
FLAP No. 2
FLAP No. 1
TORSITIONAL STIFFNESS x 10^7 LB. I
0
4
8
12
16
20
24
28
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
η ~ FRACTION OF STRUCTURAL SEMI SPAN ~ Y/(b'/2)

FIG. III- 10

FLAP SUPPORT LOADS
AT QUARTER CHORD.

SECTION	FLAP NO.	F_N INBOARD LB.(LIM.)	F_N OUTB'D. LB.(LIM.)	M INBOARD IN-LB LIMIT	M OUTB'D IN-LB LIMIT
INBOARD	1	3826	3473	-17104	-14784
	2	5077	4558	-30292	-26542
	3	2325	2124	-13204	-10907
CENTER	1	3412	3179	-10750	-9350
	2	3440	3164	-19065	-16275
	3	1572	1480	-3095	-7170
OUTB'D	1	1679	1540	-6293	-5248
	2	2153	2019	-10965	-9155
	3	1004	935	-4676	-3934

NOTE: INBOARD & OUTBOARD LOADS REFER TO LOADS AT INBOARD & OUTBOARD SUPPORTS.



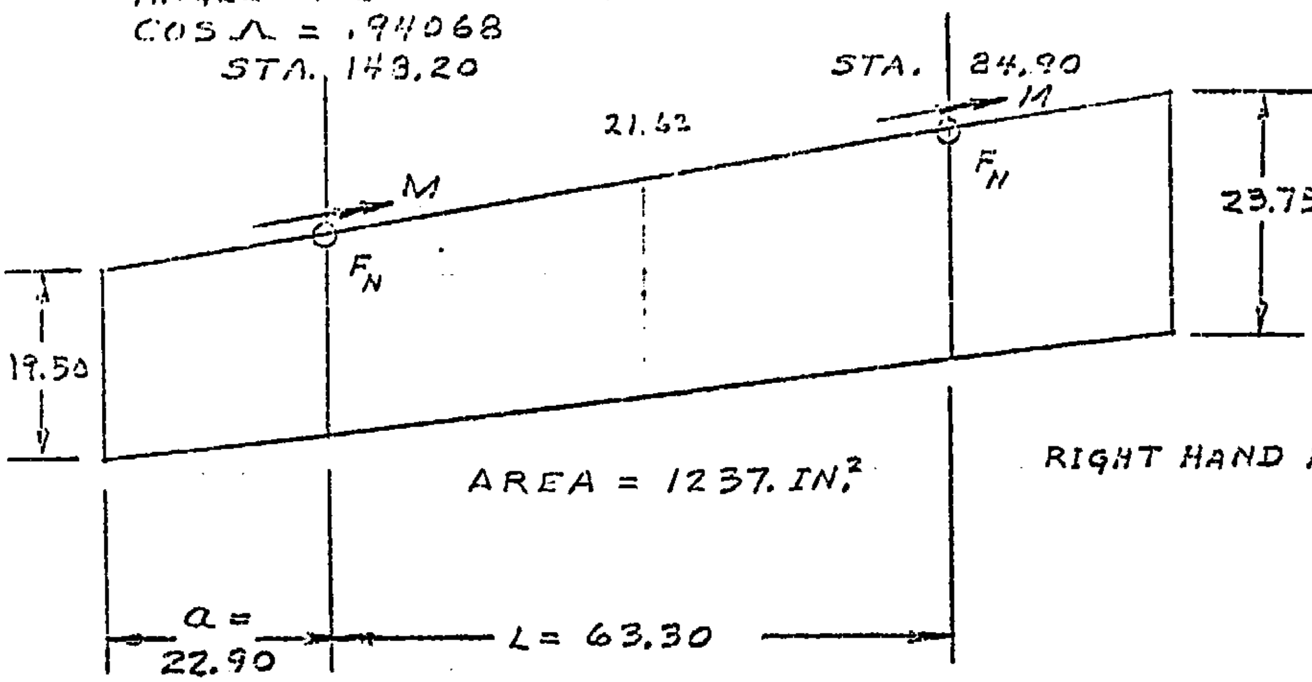
POSITIVE LOADS AS SHOWN

NORMAL FORCES AND MOMENTS AT EACH SUPPORT WERE DETERMINED BY ASSUMING AN AVERAGE OF THE DISTRIBUTION (SEE LOAD SECTION) OVER HALF OF THE FLAP LENGTH.

TABLE III-6 STOL - TRAILING EDGE FLAP LOADS

FLAP NO. 1
INBOARD

ANGLE OF SWEEP 19°50
COS Λ = .94068



RIGHT HAND RULE

$$\frac{109.12}{.94068} = 115.99 \qquad \frac{22.90}{.94068} = 24.34 \qquad \frac{63.30}{.94068} = 67.29$$

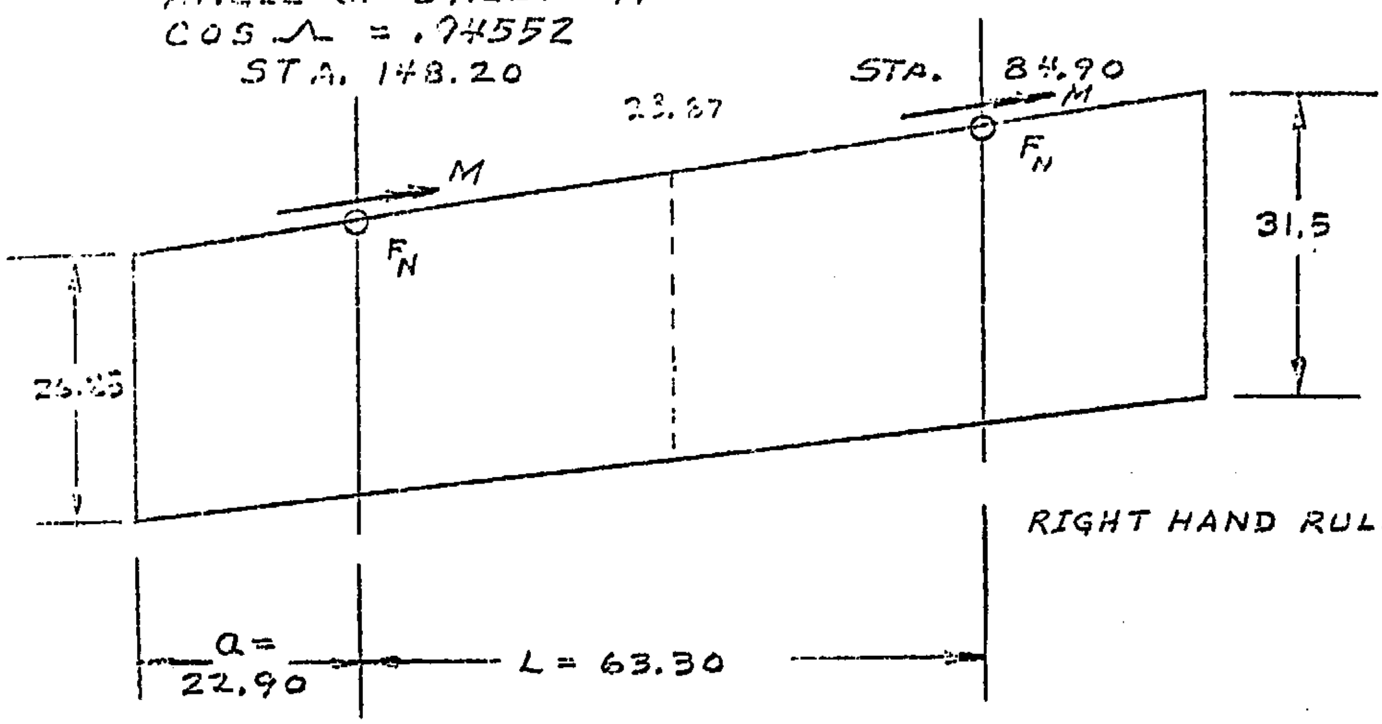
$$\frac{54.55}{.94068} = 57.98 \text{ IN.}$$

F_N = 57.98 × 60 LB./IN = 3478 LB. (LIMIT) OUTB'D SUPPORT
F_N = 57.98 × 66 LB./IN. = 3826. LB. (LIMIT) INBOARD SUPPORT
M = 57.98 × -295 = -17,104 IN-LB. (LIMIT) INBOARD SUPPORT
M = 57.98 × -255 = -14,784 IN-LB. (LIMIT) OUTB'D SUPPORT

STOL - TRAILING EDGE
INBOARD FLAP

FLAP NO. 2
INBOARD

ANGLE OF SWEEP 19°
COS Λ = .94552



$\frac{109.12}{.94552} = 115.40$ $\frac{22.90}{.94552} = 24.22$ $\frac{63.30}{.94552} = 66.75$

$\frac{54.55}{.94552} = 57.70$

30.18 X 54.55 = 1646. SQ. IN.

F_N = 57.70 X 88 LB./IN. = 5077. LB. (LIMIT) INB'D SUPPORT

F_N = 57.70 X 79 LB./IN = 4558 LB. (LIMIT) OUTB'D. SUPPORT

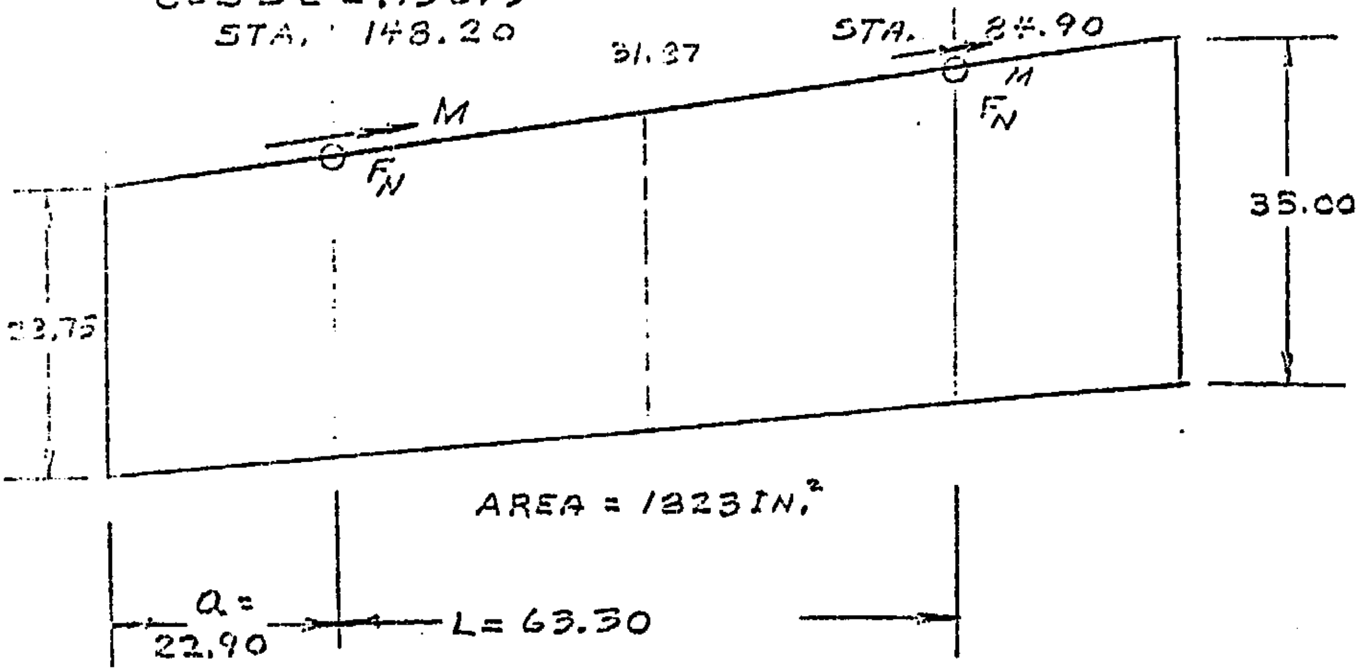
M = 57.70 X -525 = -30,292 IN-LB. (LIMIT) INB'D. SUPPORT

M = 57.70 X -460 = -26,542 IN-LB. (LIMIT) OUTB'D SUPPORT

STOL - TRAILING EDGE
INBOARD FLAP

... NO. 3
INBOARD

ANGLE OF SWEEP 18°10'
COS Λ = .95015



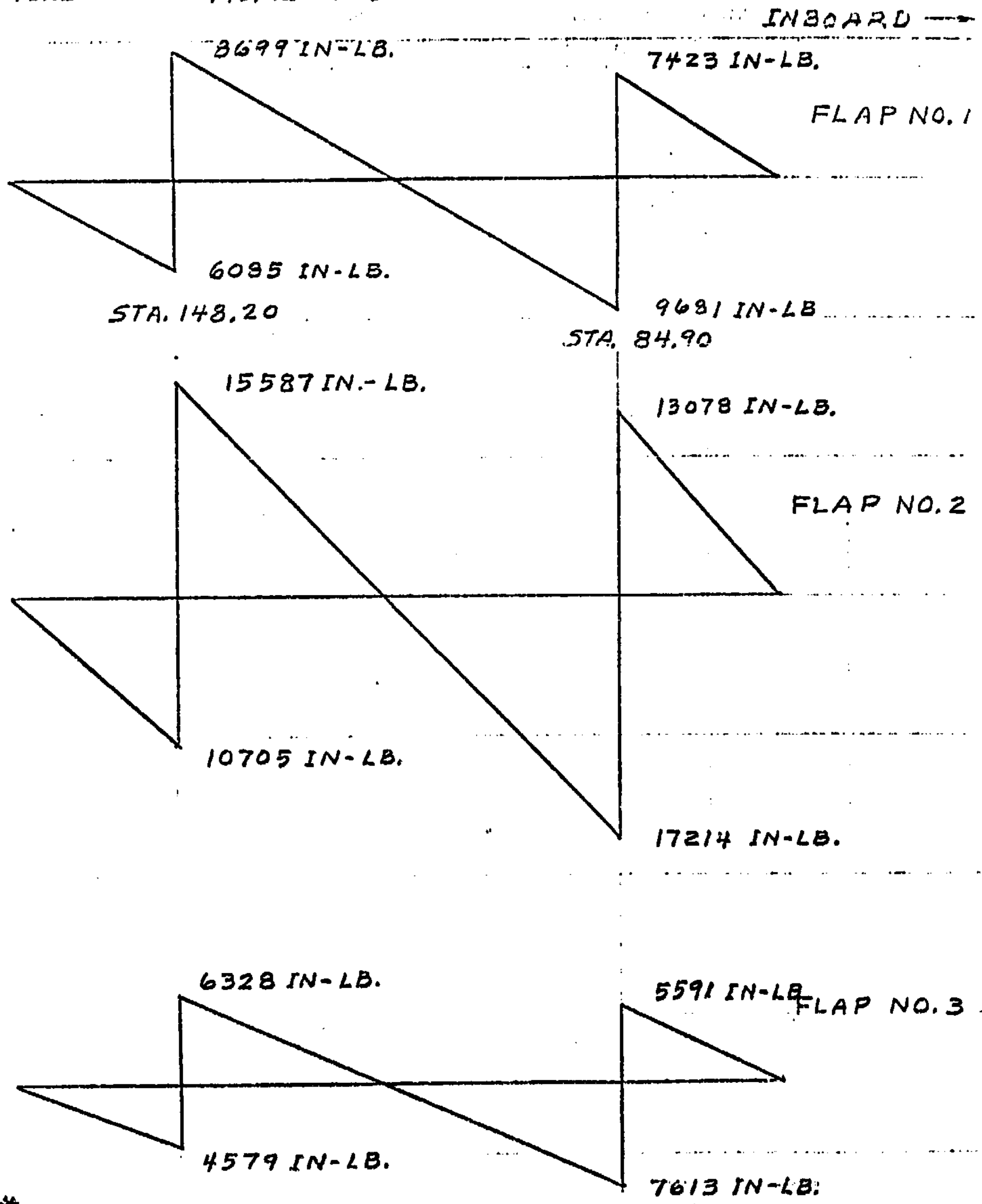
$$\frac{109.12}{.95015} = 114.85 \qquad \frac{22.90}{.95015} = 24.10 \qquad \frac{63.30}{.95015} = 66.62$$

$$\frac{54.55}{.95015} = 57.41$$

F_N = 57.41 × 37.0 = 2124. LB. (LIMIT) OUTBOARD SUPPORT
F_N = 57.41 × 40.5 = 2325. LB. (LIMIT) INBOARD SUPPORT
M = 57.41 × -230 = -13,204 IN-LB. (LIMIT) INB'D SUPPORT
M = 57.41 × -190 = - 10,907 IN-LB. (LIMIT) OUTB'D SUPPORT

STOL - TRAILING EDGE
INBOARD FLAP

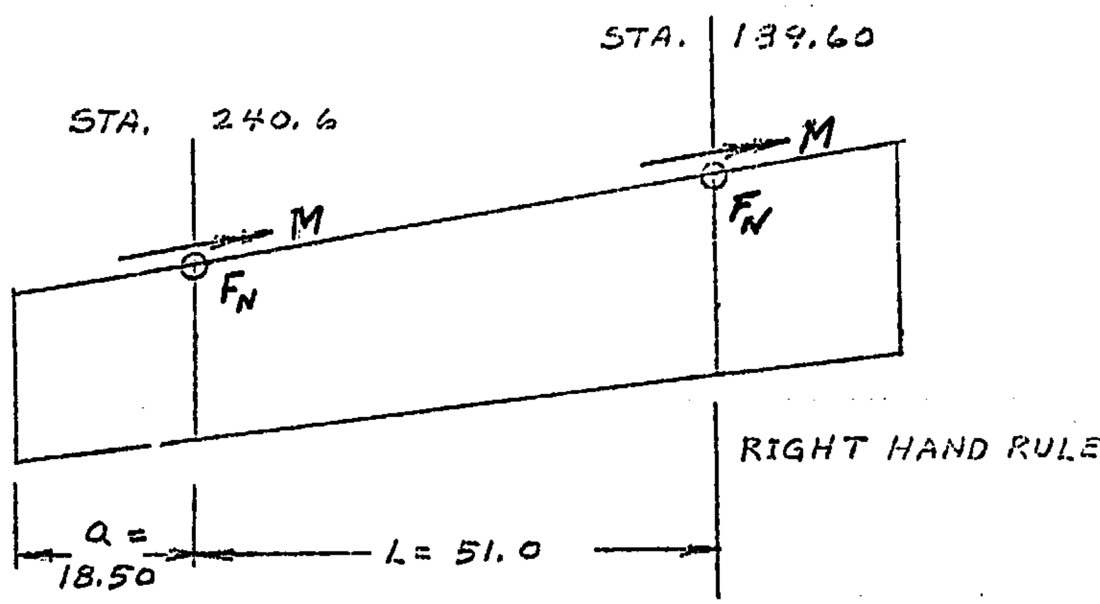
INBOARD FLAP
TORSIONAL MOMENT DISTRIBUTION*



* NOTE: TORSIONAL MOMENT DISTRIBUTION IS ASSUMED LINEAR FOR SIMPLIFICATION.

STOL - TRAILING EDGE
INBOARD FLAP

[illegible] NO. 1
CENTER



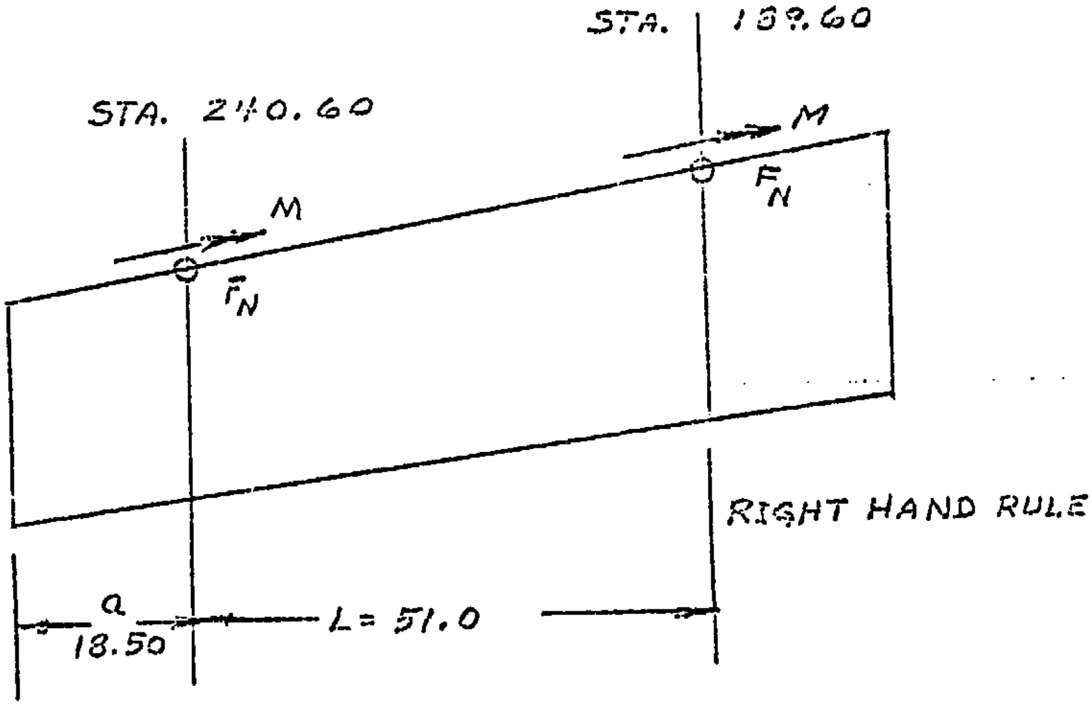
$$\frac{87.95}{.94068} = 93.496 \qquad \frac{18.50}{.94068} = 19.67 \qquad \frac{51.0}{.94068} = 54.21$$

$F_N = 54 \times 93.496 = 5050$ LB. (LIMIT) TOTAL
$F_N = 3412$ LB. (LIMIT) INB'D. $F_N = 3179$ LB. (LIMIT) OUTB'D SUPPORT

$M = -10752$ IN-LB (LIMIT) INB'D
$M = -9350$ IN-LB. (LIMIT) OUTB'D.

STOL - TRAILING EDGE
CENTER FLAP

FLAP NO. 2
CENTER



$\frac{87.95}{.94552} = 93.016 \qquad \frac{18.50}{.94552} = 19.56 \qquad \frac{51.0}{.94552} = 53.94$

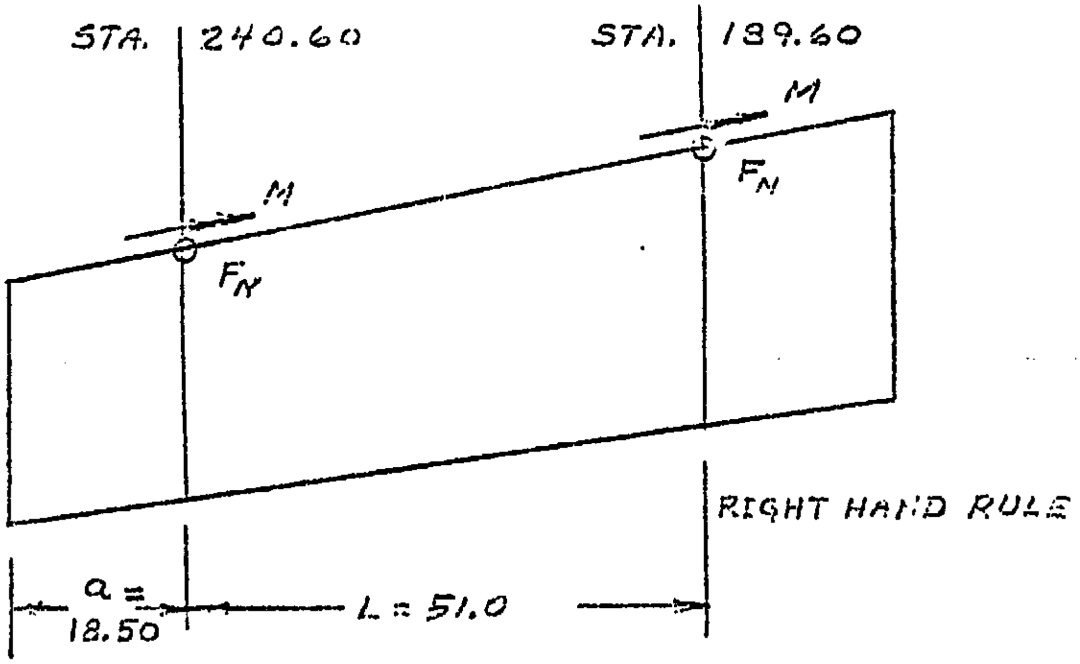
$F_N = 71.0 \times 93.016 = 6604$ LB. (LIMIT) TOTAL

$F_N = 3440$ LB. (LIMIT) INB'D
$F_N = 3164$ LB. (LIMIT) OUTB'D.

$M = -19065$ IN-LB (LIMIT) INB'D.
$M = -16275$ IN-LB (LIMIT) OUT'D.

STOL - TRAILING EDGE
CENTER FLAP

FLAP NO. 2
CENTER



$\frac{87.95}{.95015} = 92.52$ $\frac{18.50}{.95015} = 19.47$ $\frac{51.00}{.95015} = 53.67$

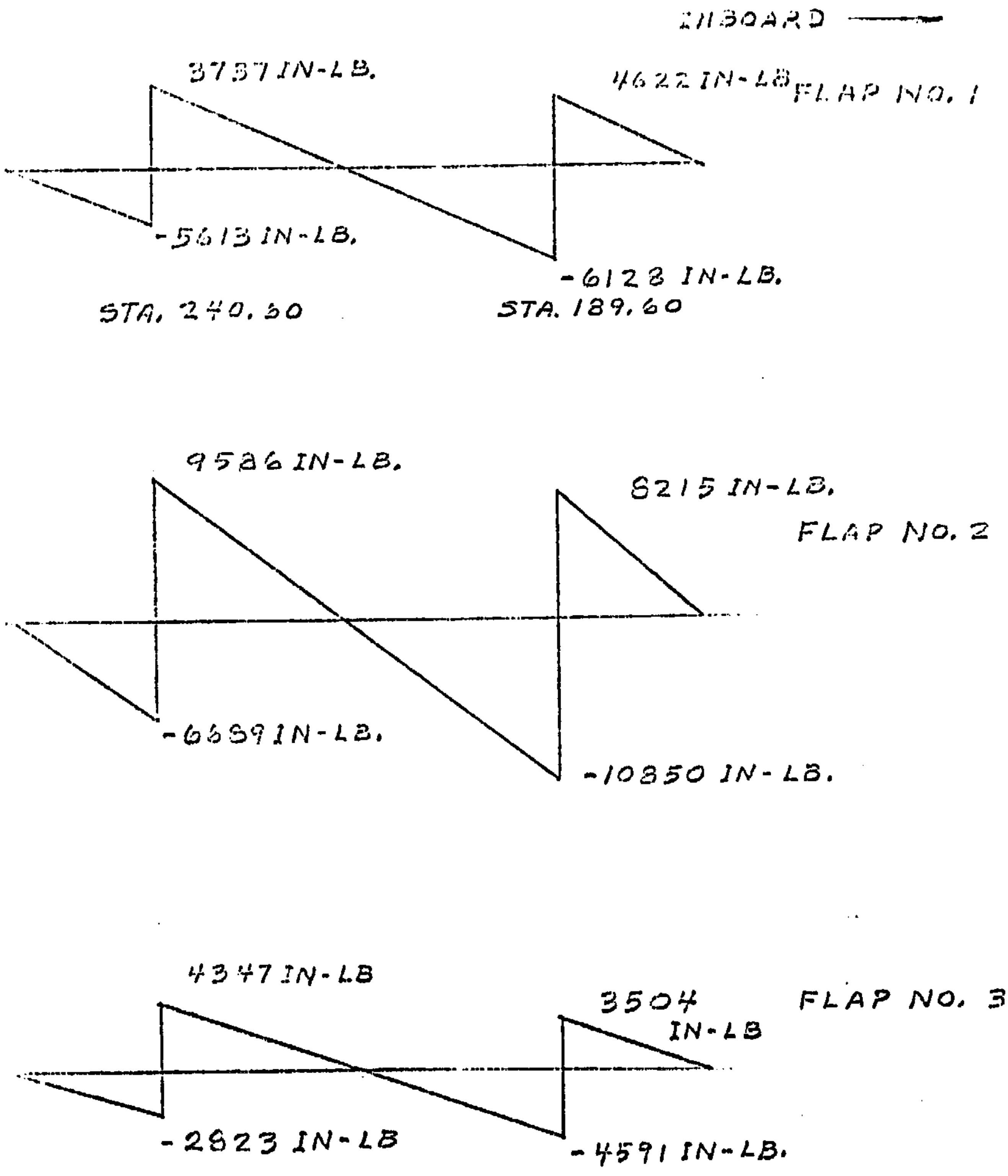
F_N = 1572. LB. (LIMIT) INB'D. F_N = 1430. LB. (LIMIT) OUTB'D

F_N = 33. × 92.52 = 3053 LB. (LIMIT) TOTAL

M = − 8095. IN-LB. (LIMIT) INB'D.
M = − 7170. IN-LB. (LIMIT) OUTB'D.

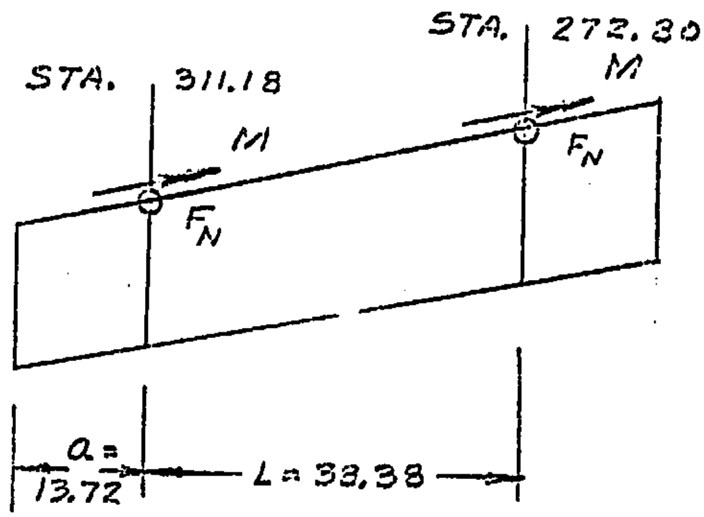
STOL - TRAILING EDGE
CENTER FLAP

[illegible] FLAP
[illegible]SIONAL MOMENT DISTRIBUTION



STOL - TRAILING EDGE
CENTER FLAP

LAP NO. 1
OUTER



$$\frac{65.83}{.94068} = 69.98 \qquad \frac{13.72}{.94068} = 14.58 \qquad \frac{38.38}{.94068} = 40.80$$

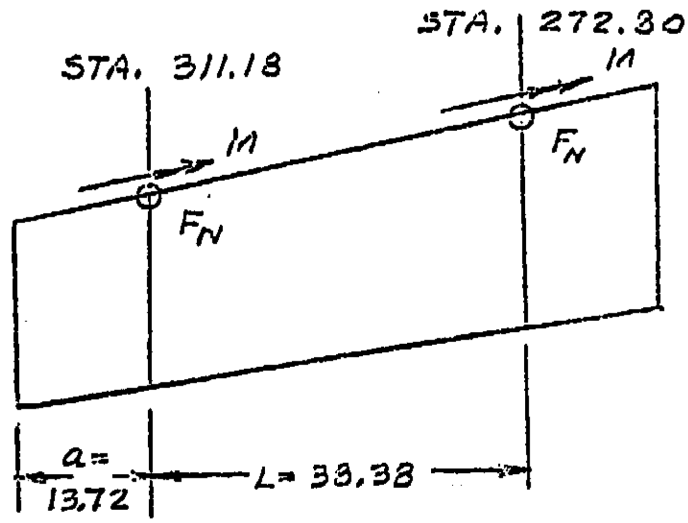
F_N = 46 × 69.93 = 3219 LB. (LIMIT) TOTAL

F_N = 1679 LB. (LIMIT) INB'D.
F_N = 1540 LB. (LIMIT) OUTB'D.

M = − 6298 IN-LB. (LIMIT) INB'D.
M = − 5243 IN-LB. (LIMIT) OUTB'D.

STOL - TRAILING EDGE
OUTER FLAP

FLAP NO. 2
OUTER



$$\frac{65.83}{.94552} = 69.62 \qquad \frac{13.72}{.94552} = 14.51 \qquad \frac{38.38}{.94552} = 40.59$$

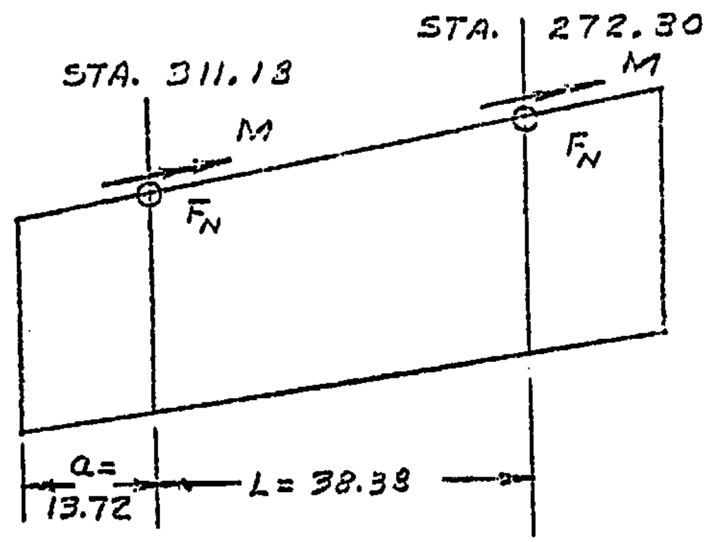
$F_N = 60 \times 69.62 = 4177$ LB. (LIMIT) TOTAL

$F_N = 2158$ LB. (LIMIT) INB'D.
$F_N = 2019$ LB. (LIMIT) OUTB'D.

$M = -10965$ IN-LB. (LIMIT) INB'D.
$M = -9155$ IN-LB (LIMIT) OUTB'D.

STOL - TRAILING EDGE
OUTER FLAP

[illegible] NO. 3
OUTER



$$\frac{65.83}{.95015} = 69.28 \qquad \frac{13.72}{.95015} = 14.44 \qquad \frac{38.38}{.95015} = 40.39$$

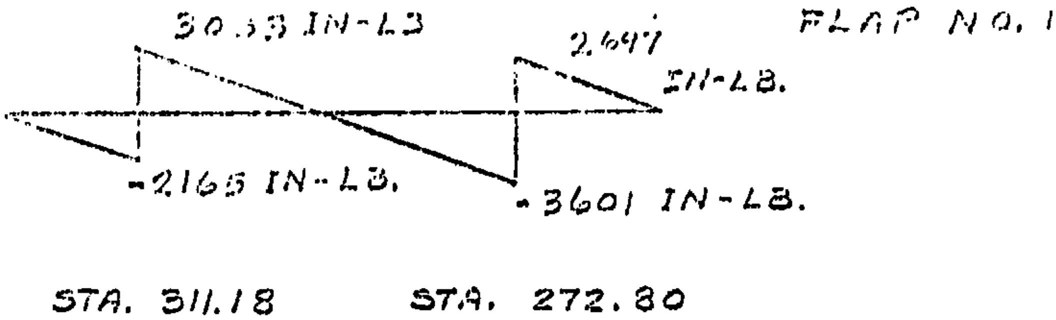
F_N = 1004 LB. (LIMIT) INB'D.
F_N = 935 LB. (LIMIT) OUTB'D.
F_N = 28. × 69.28 = 1939 LB. (LIMIT) TOTAL

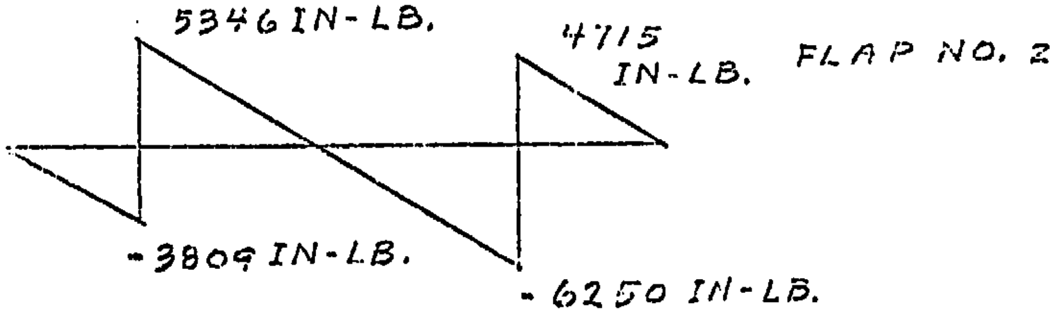
M = −4676 IN-LB. (LIMIT) INB'D.
M = −3984 IN-LB. (LIMIT) OUTB'D.

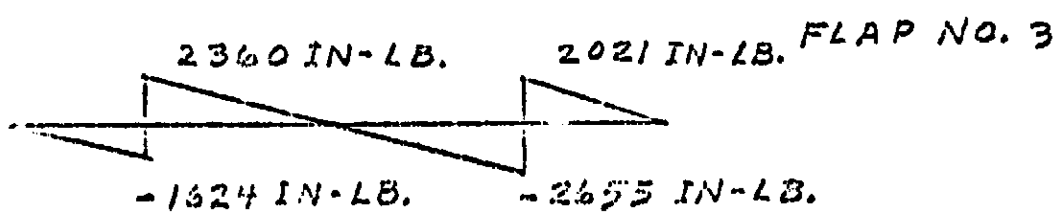
STOL - TRAILING EDGE
OUTER FLAP

OUTER FLAP
TORSIONAL MOMENT DISTRIBUTION

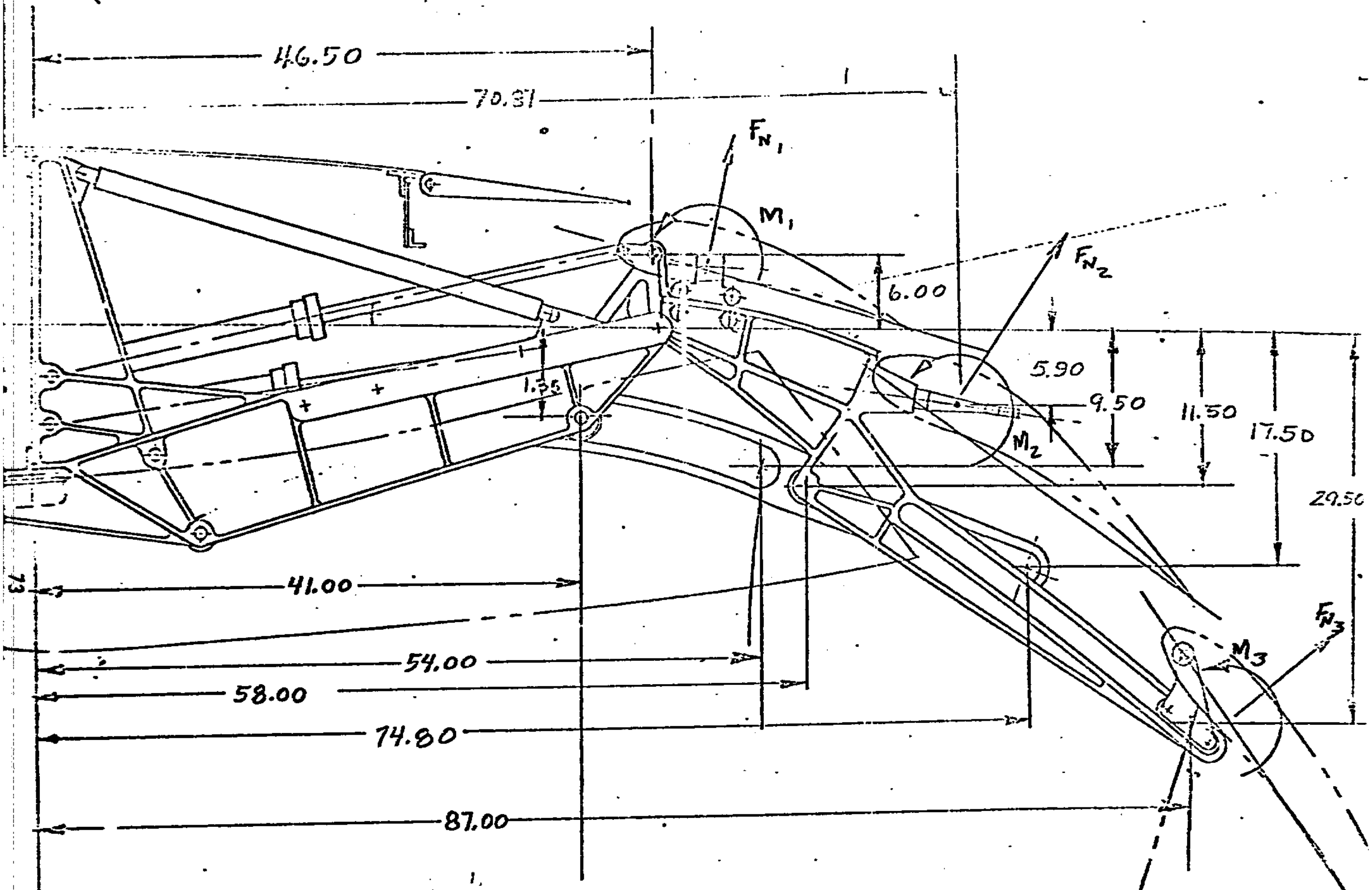
INBOARD →







STOL - TRAILING EDGE
OUTER FLAP



INBOARD FLAP

FIG III - 11

[illegible]BOARD FLAP

F_N (INB'D. SUPPORT)	M (INB'D SUPPORT)
2325 LB. FLAP NO. 3	−13,204 IN-LB. (LIMIT)
5077 LB. FLAP NO. 2	−30,292 IN-LB.
3826 LB. FLAP NO. 1	−17,104 IN-LB.

TAKING MOMENTS ABOUT THE PIVOT POINT ON FLAP NO. 1

$$\Sigma M = 0 = 17.7\,R_B - 43.5 \times 2325 \cos 22^\circ - 14.3 \times 5077 \cos 42^\circ$$
$$- 1.0 \times 3826 \cos 63^\circ - 32.0 \times 2325 \sin 22^\circ$$
$$- 16.0 \times 5077 \sin 42^\circ - 1.6 \times 3826 \sin 63^\circ$$
$$- 19249 - 55677 - 17104 = 0$$

$$17.7\,R_B - 323{,}926 = 0$$
$$R_B = 18{,}583 \text{ LB. (ROLLER LOAD)}$$

$$18{,}335 \sin 54^\circ = 14700 \text{ LB.}$$
$$18{,}335 \cos 54^\circ = 10800 \text{ LB.}$$

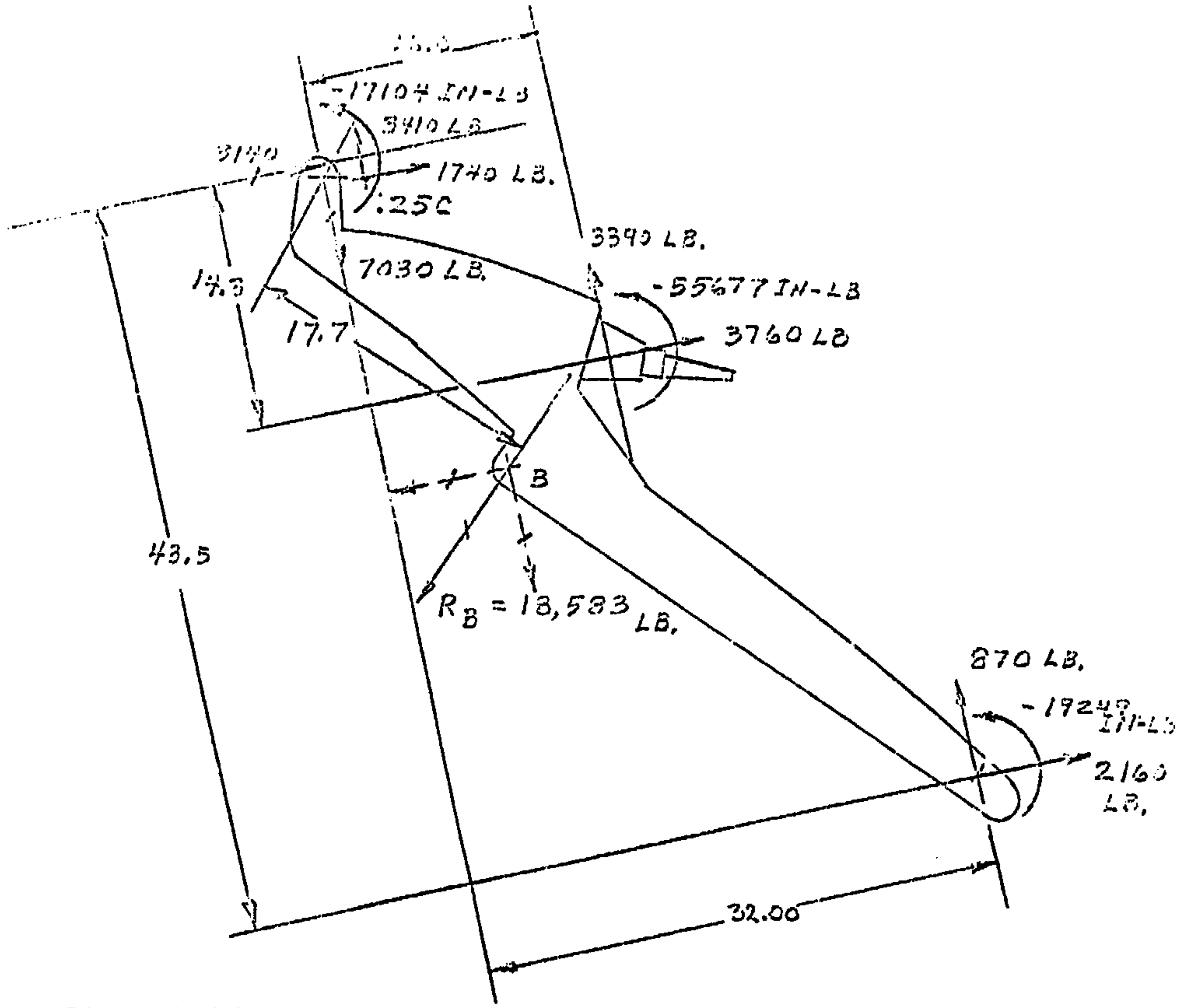
ΣF_H	ΣF_V
3760	3410
1740	3390
2160	870
7660	7670

$10{,}880 - 7660 = 3140$ ACTUATOR LOAD

$14{,}700 - 7670 = 7030$ REACTION AT TRI-POD

F_N & M = FORCE & MOMENT PER SUPPORT.

FOR SIMPLICITY, THE MOMENTS WERE USED DIRECTLY IN THE EQUATIONS OF EQUILIBRIUM WITHOUT CONSIDERATION THAT THIS PLANE IS NOT NORMAL TO THE FLAP QUARTER CHORD (6% ERROR)



DIMENSIONS SHOWN ARE THOSE FOR THE INBOARD SUPPORT.

MOMENTS SHOWN RESULT FROM TRANSFERRING F_N ₤ M FORWARD.

STOL - TRAILING EDGE
INBOARD FLAP

INBOARD FLAPS

TAKING MOMENTS ABOUT THE LOWER REAR SPAR TO FIND HORIZONTAL REACTION AT UPPER SPAR CAP.

$$\Sigma M = 0 = 21.3\,R_U - 2325 \times 68.5 - 5077 \times 55.5$$
$$- 3826 \times 44.0 - 13,204 - 30,292 - 17,104$$

$$21.3\,R_U - 159,262 - 281,775 - 168,344 - 60600 = 0$$

$$R_U = \frac{669,979}{21.3} = 31,454 \text{ LB.}$$

R_U = HORIZONTAL REACTION AT REAR SPAR (UPPER)

CHECK:

$$21.3\,R_U - 18,335 \times 54.4 + 7030 \times 48$$

$$R_U = \frac{-687,184}{21.3} = 32,262 \text{ LB.}$$

STOL - TRAILING EDGE
INBOARD FLAP

INBOARD FLAPS

TUBE

$L = 37.0$	A	I
O.D. 1.750	2.403	.4604
I.D. 1.500	1.767	.2485
	.636	.2119

$$\rho = \sqrt{\frac{I}{A}} = \sqrt{.300} = .55$$

$$\frac{L}{\rho} = \frac{37.}{.550} = 67.3$$

$F_{CR} = 65,000$ PSI

$$\frac{49,500}{.636 \times 2} = 38,900 \text{ PSI}$$

$$J = \sqrt{\frac{EI}{P}} = \sqrt{\frac{30.0 \times 10^6 \times .2119}{24.75 \times 10^3}} = 16$$

$$\frac{L}{J} = \frac{37}{16} = 2.31$$

$$\frac{M}{M_1} = 2.7$$

$M_1 = 2475$ IN-LB.

$M = 6682$ IN-LB

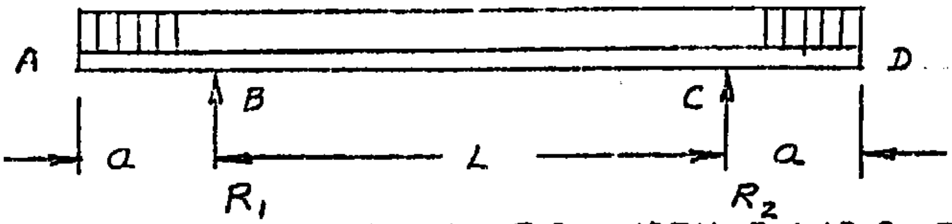
$$f = \frac{6682 \times .875}{.2119} = -27,592 \text{ PSI}$$

$$-38900 - 27592 = -66,500$$

DUE TO THE REDUNDANCY OF THE SUPPORT, THE TUBE IS CONSIDERED TO BE ADEQUATELY SIZED.

STOL
FLAPS
DEFLECTION ANALYSIS

THE FLAPS CONSISTS OF THREE SECTIONS: INBOARD, CENTER, AND OUTBOARD. EACH SECTION HAS THREE AIRFOIL SHAPE PARTS SUPPORTED BY TWO SETS OF TRACKS.



AD REPRESENTS THE FLAPS, WITH B AND C REPRESENTING THE TRACK SUPPORTS.

$$\Sigma M_C = 0 = R_1 L + \frac{wa^2}{2} - \frac{w}{2}(a+L)^2$$

$$R_1 = \frac{wL}{2} + wa$$

THE BENDING MOMENT BETWEEN B AND C IS

$$M_x = R_1 X - \frac{w(a+x)^2}{2} = R_1 X - \frac{wa^2}{2} - wax - \frac{wX^2}{2}$$

$$= -\frac{wX^2}{2} + \frac{wLX}{2} - \frac{wa^2}{2}$$

FOR ALL POINTS BETWEEN B AND C

$$M_X = EI\frac{d^2Y}{dX^2} = -\frac{wX^2}{2} + \frac{wLX}{2} - \frac{wa^2}{2}$$

ASSUME THAT THE MOMENT OF INERTIA IS CONSTANT, USING I_{AV}, THEN THIS EQUATION CAN BE INTEGRATED.

B.C.

$$y'\left(\frac{L}{2}\right) = 0 \quad \text{AND} \quad y(0) = 0$$

$$y = y_{MAX} = \Delta \quad \text{WHEN} \quad x = \frac{L}{2}$$

THE ELASTIC CURVE OF THE CENTRAL PORTION OF THE FLAP

$$EIy = -\frac{wx^4}{24} + \frac{wLX^3}{12} - \frac{wa^2X^2}{4} - \frac{wL^3X}{24} + \frac{wa^2LX}{4}$$

FOR MAXIMUM DEFLECTION $X = L/2$

$$EIy = \frac{-wL^4}{384} + \frac{wL^4}{96} - \frac{wa^2L^2}{16} - \frac{wL^4}{48} + \frac{wa^2L^2}{8}$$

$$= -\frac{5wL^4}{384} + \frac{wa^2L^2}{16}$$

STRESS
DEFLECTION
FLAP NO. 1 (INBOARD)

$w = 63$ LB./IN (AV.)
$a = 24.34$ IN. $\quad L = 67.29$ IN.
$I = 2.4\ IN^4$ $\quad t = .030$ IN.

$$M_X = -\frac{wX^2}{2} + \frac{wLX}{2} - \frac{wa^2}{2}$$

$$M_{X=0} = -\frac{wa^2}{2} = -\frac{63}{2}(24.34)^2 = -18,661 \text{ IN-LB.}$$

$$M_{X=\frac{L}{2}} = -31.5\,(33.64)^2 + 31.5\,(67.29)(33.64) - 18,661 = 17,061$$

THE STRESS AT THE SUPPORT

$$f_b = \frac{MC}{I} = \frac{-18,661 \times 1.55}{2.4} = \pm\ 12,051 \text{ PSI}$$

$$\frac{q}{t} = \frac{M}{2At} = \frac{9631}{2 \times 56 \times .030} = 2881 \text{ PSI}$$

MAX. DEFLECTION

$$EIy = -\frac{5wL^4}{384} + \frac{wa^2L^2}{16} = -\frac{5 \times 63 \times (67.29)^4}{384} + \frac{63 \times (24.34)^2 (67.29)^2}{16}$$

$$y = -\frac{6.27 \times 10^6}{30.0 \times 10^6 \times 2.4} = -0.0833 \text{ IN.}$$

$t = .030$ IN. $\quad E = 30.0 \times 10^6$ PSI

STRESS
DEFLECTION
FLAP NO. 2 (INBOARD)

$w = 83.5$ LB./IN. (AV.) $\qquad E = 30.0 \times 10^6$ PSI

$a = 24.21$ IN. $\qquad L = 66.94$ IN

$t = .030$ IN.

$$M_x = -\frac{wx^2}{2} + \frac{wLx}{2} - \frac{wa^2}{2}$$

$$M_{x=0} = -\frac{wa^2}{2} = -\frac{83.5}{2}(24.22)^2 = -24,197 \text{ IN-LB.}$$

$$M_{x=L/4} = -41.75(16.73)^2 + 41.75(66.95)(16.73) - 24,197$$

$$= -11545 + 46200 - 24,197 = 10,458 \text{ IN-LB.}$$

$$M_{x=L/2} = -41.75(33.47)^2 + 41.75(66.95)(33.47) - 24,197$$

$$- 46,200 + 92,400 - 24,197 = 22,003 \text{ IN-LB.}$$

$$f_b = \frac{MC}{I} = \frac{24197 \times 1.5}{4.66} = \pm 7785 \text{ PSI}$$

$$\frac{q}{t} = \frac{M}{2At} = \frac{17214}{2 \times 91 \times .030} = 3152 \text{ PSI}$$

MAX. DEFLECTION

$$EIy = -\frac{5 \times 83.5 \times (66.94)^4}{384} + \frac{83.5 \times (24.21)^2 \times (66.94)^2}{16}$$

$$y = -\frac{8.03 \times 10^6}{30.0 \times 10^6 \times 4.2} = -.0637 \text{ IN.}$$

STOL - TRAILING EDGE
INBOARD FLAPS

STRESS
DEFLECTION
FLAP NO. [illegible]

$w = 38.5$ LB/IN. (AV.) $\qquad E = 30.0 \times 10^6$ PSI

$a = 24.10$ IN. $\qquad L = 66.62$ IN.

$t = .030$

$$M_x = -\frac{wx^2}{2} + \frac{wLx}{2} - \frac{wa^2}{2}$$

$$M_{x=0} = -\frac{wa^2}{2} = -\frac{38.5}{2}(24.10)^2 = -11,180 \text{ IN-LB.}$$

$$M_{x=L/4} = -19.25\,(16.65)^2 + 19.25\,(66.62)(16.65) - 11180$$
$$= -5336 + 21,352 - 11180 = 4836 \text{ IN-LB.}$$

$$M_{x=L/2} = -19.25\,(33.31)^2 + 19.25\,(66.62)(33.31) - 11180$$
$$-21358 + 42,717 - 11180 = 10,179 \text{ IN-LB}$$

$$f_b = \frac{Mc}{I} = \frac{11,180 \times 1.60}{3.28} = 5,453 \text{ PSI}$$

$$\frac{q}{t} = \frac{T}{2At} = \frac{7613}{2 \times 103 \times .030} = 1230 \text{ PSI}$$

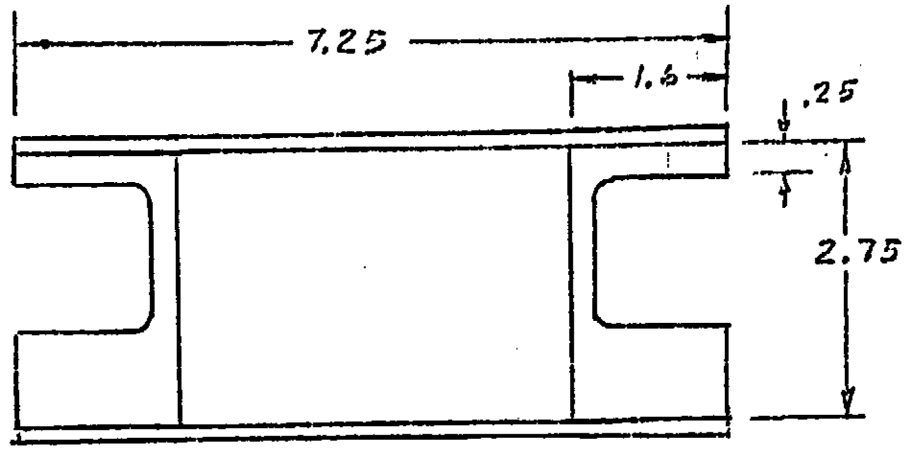
MAX. DEFLECTION

$$EIy = -\frac{5 \times 38.5 \times (66.62)^4}{384} + \frac{38.5 \times (24.10)^2 \times (66.62)^2}{16}$$

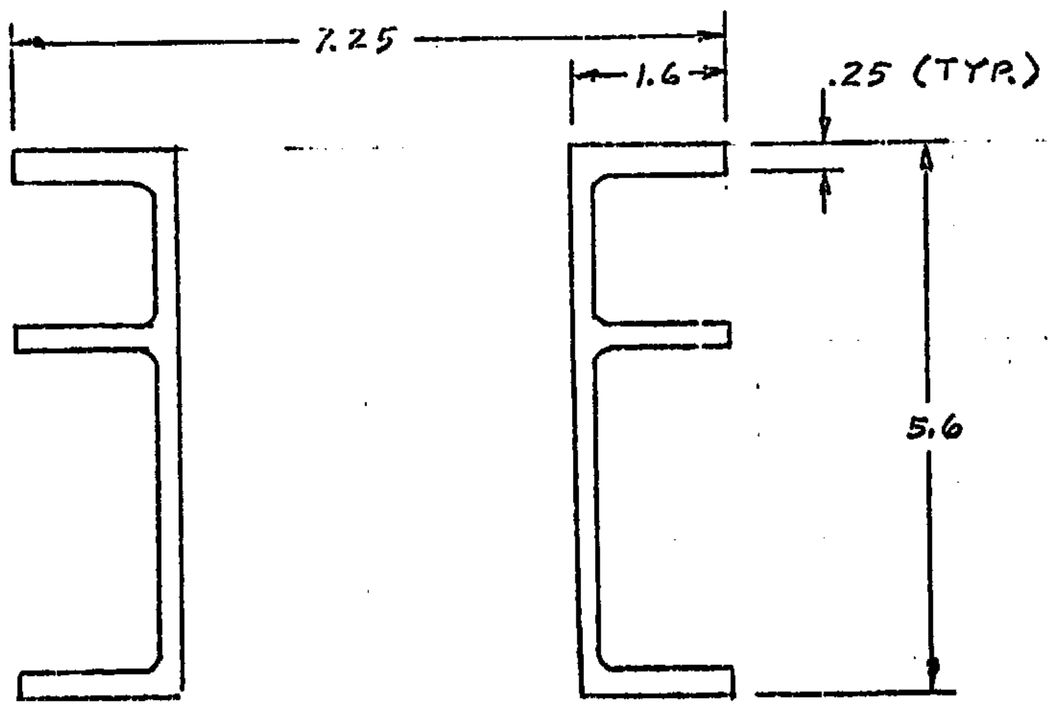
$$y = -\frac{3,672 \times 10^6}{30.0 \times 10^6 \times 4.92} = -.0249 \text{ IN.}$$

STOL - TRAILING EDGE
INBOARD FLAP

INBOARD FLAP
BEAM



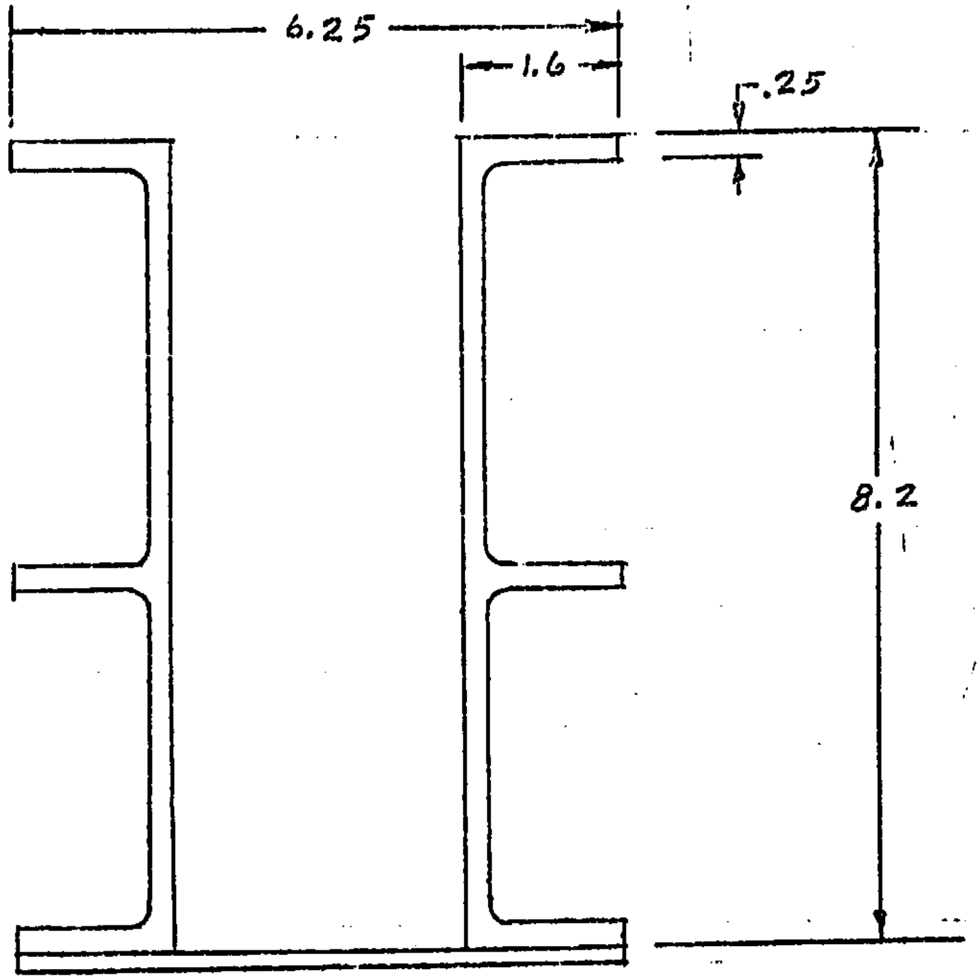
A - A
AREA = $5.475\ IN.^2$
I_{YY} = $3.940\ IN.^4$
I_{ZZ} = $16.92\ IN.^4$



B - B
AREA = $5.135\ IN.^2$
I_{YY} = $17.80\ IN.^4$
I_{ZZ} = $21.80\ IN.^4$

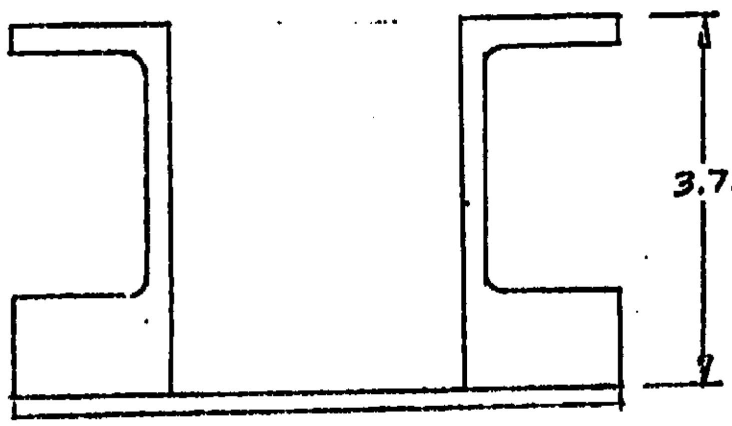
STOL. - TRAILING EDGE
INBOARD FLAP

INBOARD FLAP
BEAM



CC

AREA = 6.067 $IN.^2$
I_{YY} = 47.90 $IN.^4$
I_{ZZ} = 20.32 $IN.^4$

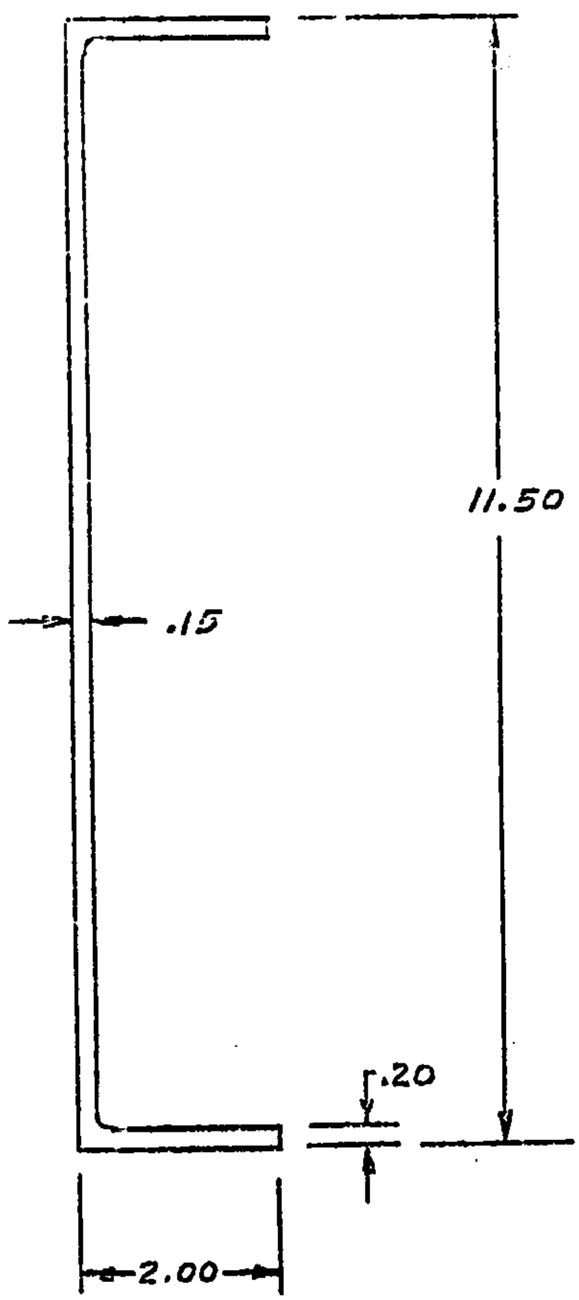


AREA = 5.20 $IN.^2$
I_{YY} = 5.97 $IN.^4$
I_{ZZ} = 13.30 $IN.^4$

D - D

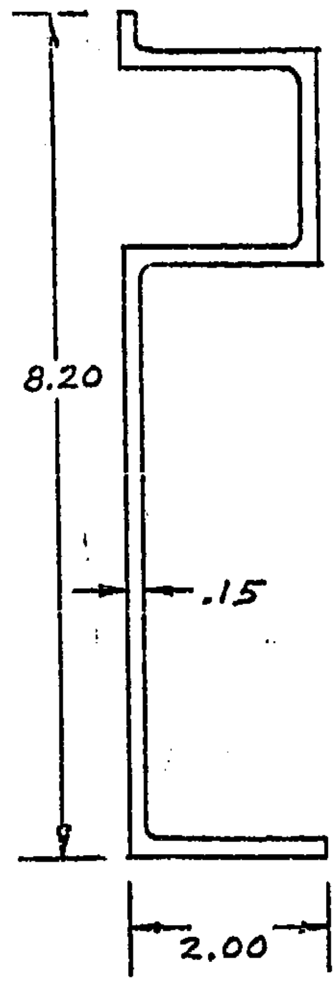
STOL - TRAILING EDGE
INBOARD FLAP

INBOARD FLAP
BEAM



A = 2.49 IN^2
I = 43.53 $IN.^4$

H - H



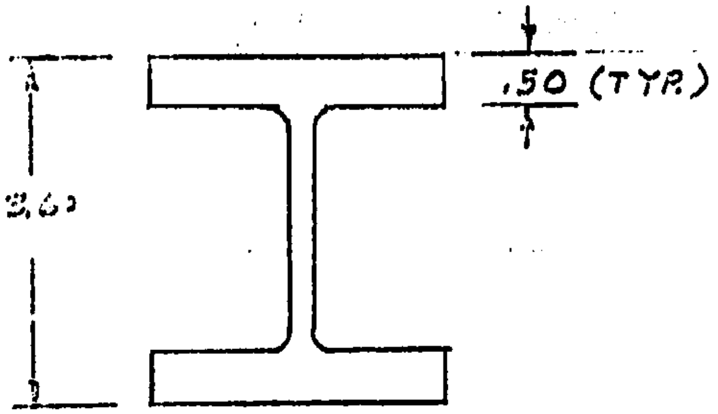
J - J

A = 1.61 $IN.^2$
I = 8.24 $IN.^4$

EACH OF THESE SECTIONS ARE SYMMETRICAL WITH RESPECT TO A VERTICAL ℄

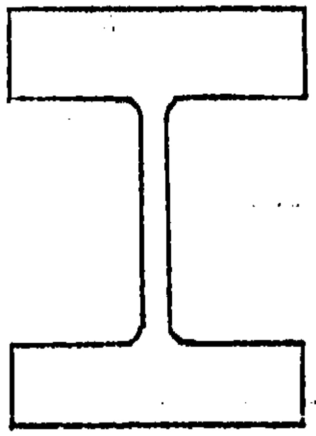
STOL - TRAILING EDGE
INBOARD FLAP

INBOARD FLAP TRACK



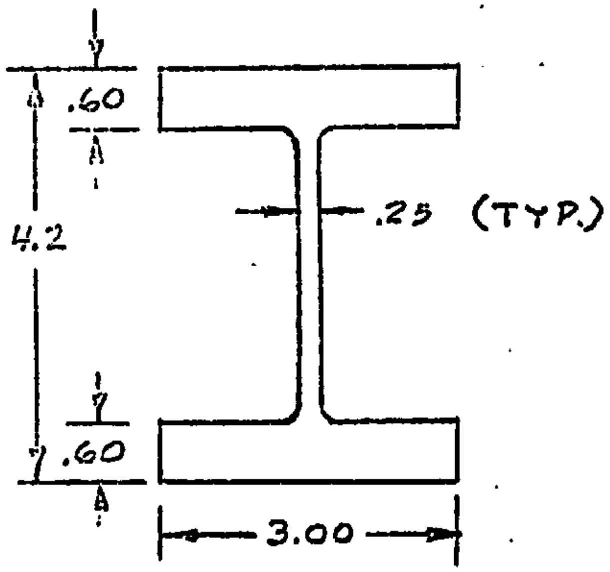
E-E

A = 3.62 IN.^{2}
I_{YY} = 7.07 IN.^{4}
I_{ZZ} = 2.25 IN.^{4}



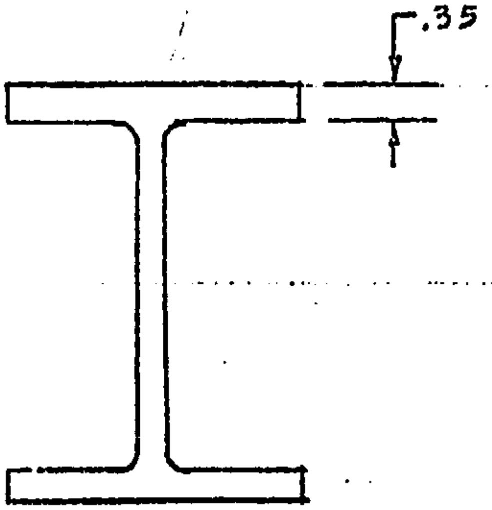
F-F *

A = 5.42
I_{YY} = 13.38 IN.^{4}
I_{ZZ} = 3.60 IN.^{4}



G-G

A = 4.50 IN.^{2}
I_{YY} = 12.56 IN.^{4}
I_{ZZ} = 2.70 IN.^{4}

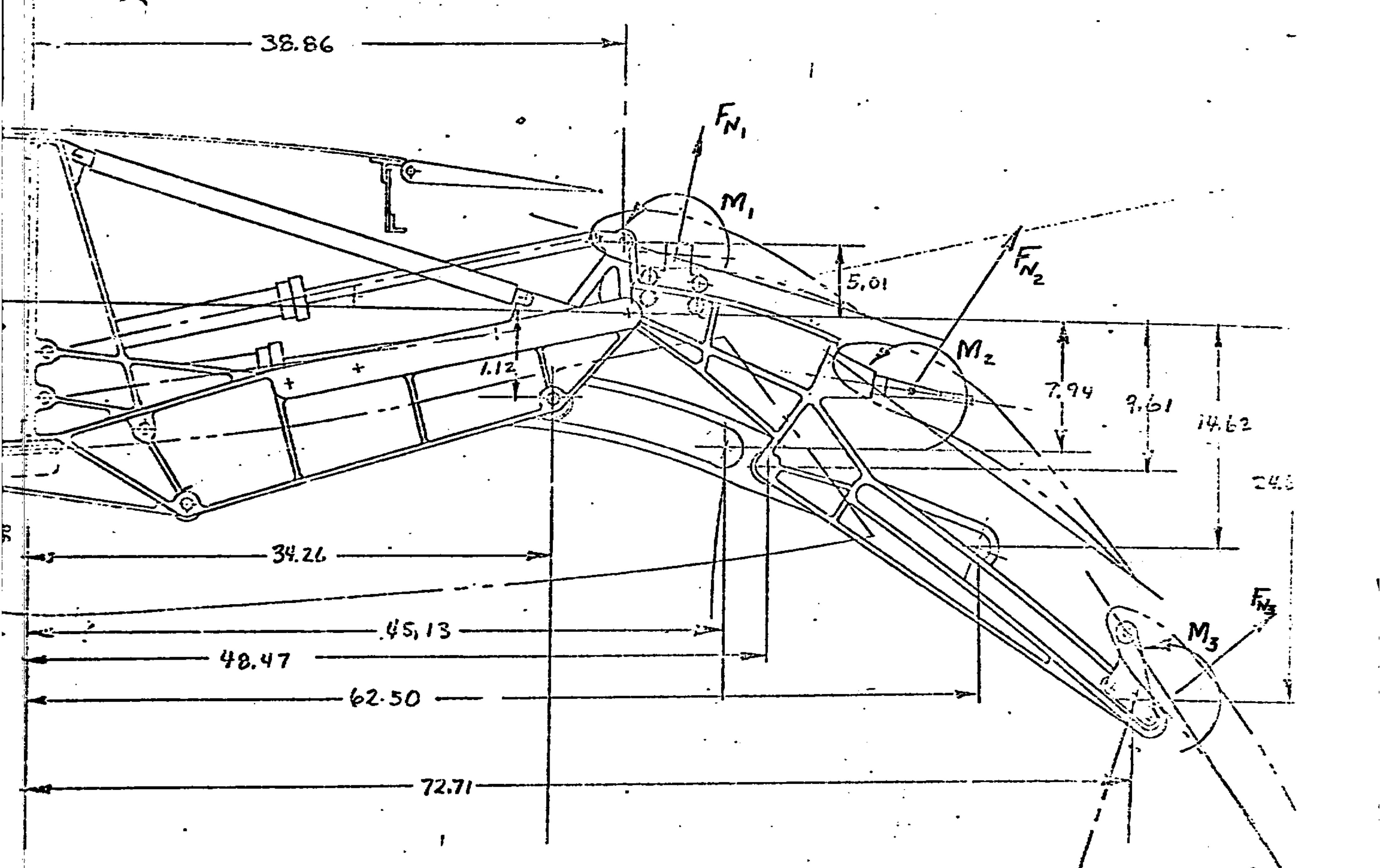


I-I

A = 2.97 IN.^{2}
I_{YY} = 8.71 IN.^{4}
I_{ZZ} = 1.57 IN.^{4}

* SECTION F-F IS IN A TRANSITION REGION.

STOL - TRAILING EDGE
INBOARD FLAP


38.86
F_{N_1}
M_1
5.01
F_{N_2}
M_2
1.12
7.94
9.61
14.62
34.26
45.13
48.47
62.50
72.71
F_{N_3}
M_3

CENTER FLAP

CNTER FLAP

F_N (TOTAL LOAD)		M (TOTAL MOMENT)
3053	FLAP NO. 3	-15,265 IN-LB. (LIMIT)
6604	FLAP NO. 2	-35,346 IN-LB.
5050	FLAP NO. 1	-20,100 IN-LB.

TAKING MOMENTS ABOUT THE PIVOT POINT ON FLAP NO. 1 TO FIND HORIZONTAL & VERTICAL COMPONENTS OF THE REACTION AT B

$$\Sigma M = 0 = 9.66 R_{B_V} + H_1 \times 1526 \sin 75^\circ + H_2 \times 3302 \sin 55^\circ + H_3 \times 2525 \sin 35^\circ$$

$$9.66 R_{B_V} + 2.50 \times 1470 + 19.02 \times 2700 + 36.57 \times 1450 = 0$$

$$R_{B_V} = \frac{108055}{9.66} = 11{,}185 \text{ LB.}$$

$$10.94 R_{B_H} + V_1 \times 1526 \cos 75^\circ + V_2 \times 3302 \cos 55^\circ + V_3 \times 2525 \cos 35^\circ = 0$$

$$10.94 R_{B_H} + .3 \times 395 \qquad + 14.45 \times 23.25 + 29.53 \times 2070 = 0$$

$$R_{B_H} = \frac{95142}{10.94} = 8696 \text{ LB.}$$

$$R_B = \left\{(11,135)^2 + (8696)^2\right\}^{1/2} + \frac{70711}{2 \times 14.63} = 16,570 \text{ LB.}$$

STOL - TRAILING EDGE
CENTER FLAPS

FLAP NO. 1 (CENTER)

$$M_x = -\frac{\omega x^2}{2} + \frac{\omega L x}{2} - \frac{\omega a^2}{2}$$

$$M_{x=0} = -\frac{\omega a^2}{2} = -\frac{52}{2}(19.67)^2 = -10059 \text{ IN.-LB.}$$

$\omega = 52$ LB/IN. $\quad a = 19.67$ IN. $\quad L = 54.21$ IN. $\quad t = .025$

$$f_b = \frac{MC}{I} = \frac{-10059 \times 1.05}{2.6} = \pm 4062 \text{ PSI}$$

$$\frac{q}{t} = \frac{T}{2At} = \frac{6128}{2 \times 39 \times .025} = 3142 \text{ PSI}$$

$y = -.0648$ IN. $\quad E = 30.0 \times 10^6$ PSI

FLAP NO. 2 (CENTER)

$$M_{x=0} = -\frac{\omega a^2}{2} = \frac{-70 \times (19.67)^2}{2} = -13541 \text{ IN-LB.}$$

$\omega = 70.$ LB./IN.

$$f_b \quad \frac{MC}{I} = \frac{13541 \times 1.2}{2.1} = \pm 7737 \text{ PSI}$$

$$\frac{q}{t} = \frac{T}{2At} = \frac{10850}{2 \times 65 \times .025} = 3,338 \text{ PSI}$$

$y = -.046$ IN. $\quad E = 30.0 \times 10^6$ PSI

FLAP NO. 3 (CENTER)

$$M_{x=0} = -\frac{\omega a^2}{2} = -\frac{33 \times (19.67)^2}{2} = -6384 \text{ IN-LB.}$$

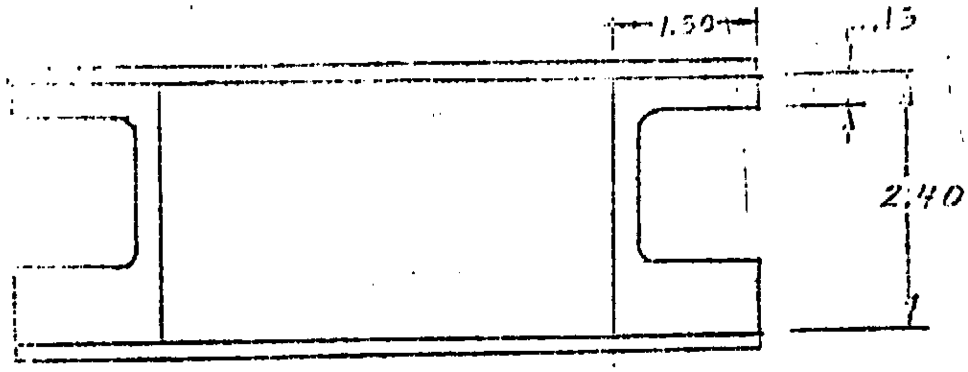
$\omega = 33.$ LB/IN.

$$f_b = \frac{MC}{I} = \frac{6384 \times 1.31}{2.4} = \pm 3492 \text{ PSI}$$

$$\frac{q}{t} = \frac{T}{2At} = \frac{4591}{2 \times 75 \times .025} = 1224 \text{ PSI}$$

$y = -.019$ IN. $\quad E = 30.0 \times 10^6$ PSI

CENTER FLAP - BEAM

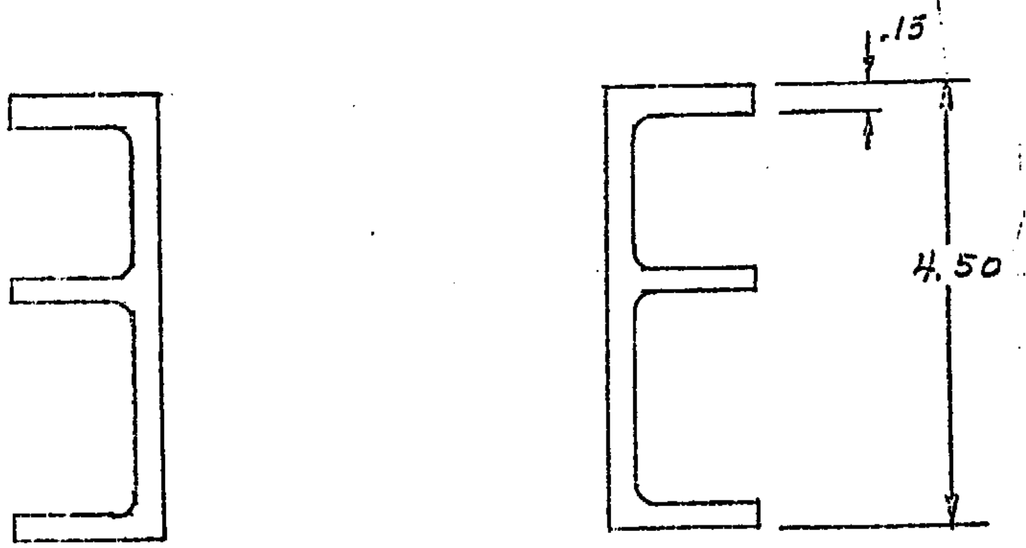


A - A

AREA = 3.90 $IN.^2$

I_{YY} = 2.90 $IN.^4$

I_{ZZ} = 23.43 $IN.^4$



B - B

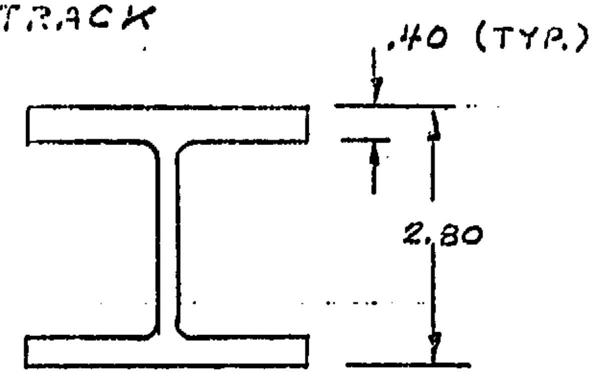
AREA = 3.35 $IN.^2$

I_{YY} = 7.63 $IN.^4$

I_{ZZ} = 22.45 $IN.^4$

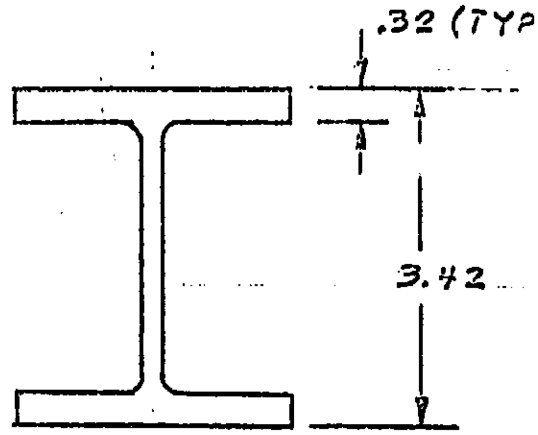
STOL - TRAILING EDGE
CENTER FLAP

CENTER FLAP
TRACK



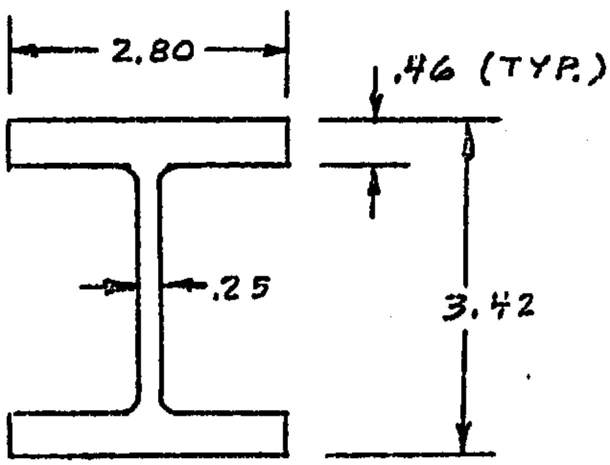
E - E

A = 2.54 $IN.^2$
I_{YY} = 3.36 $IN.^4$
I_{ZZ} = 1.46 $IN.^4$



I - I

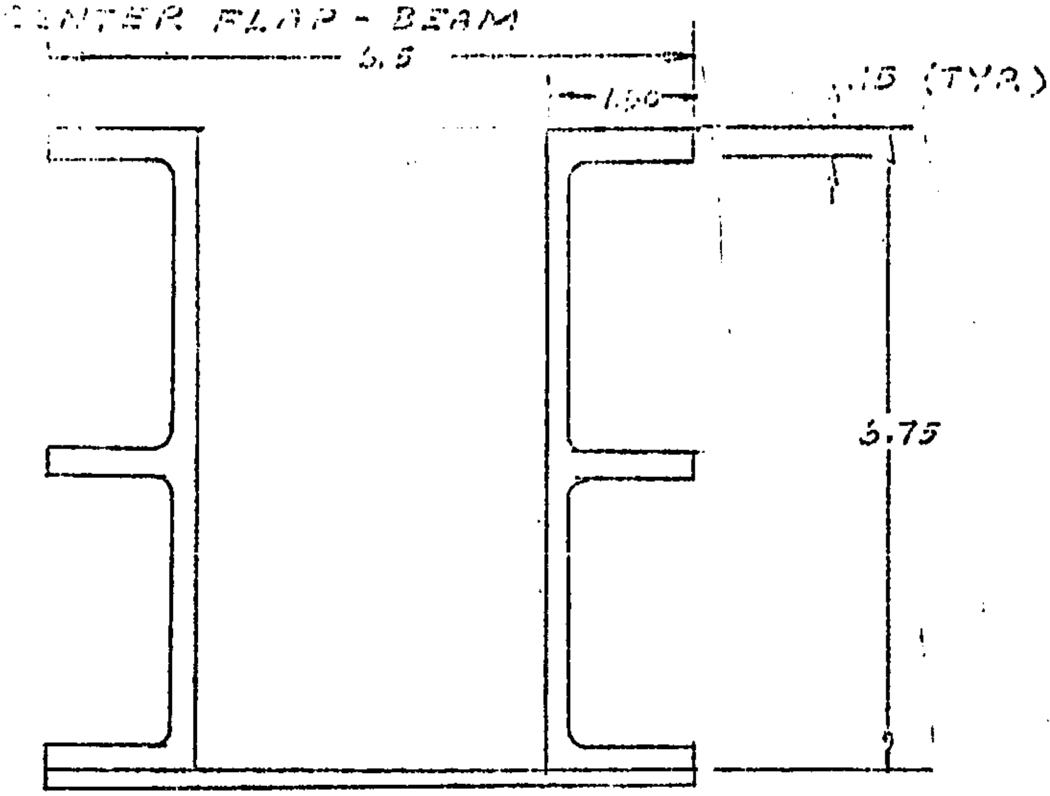
A = 2.42
I_{YY} = 2.66 $IN.^4$
I_{ZZ} = 1.17 $IN.^4$



G - G

A = 3.07 $IN.^2$
I_{YY} = 5.84 $IN.^4$
I_{ZZ} = 1.68 $IN.^4$

STOL - TRAILING EDGE
CENTER FLAP

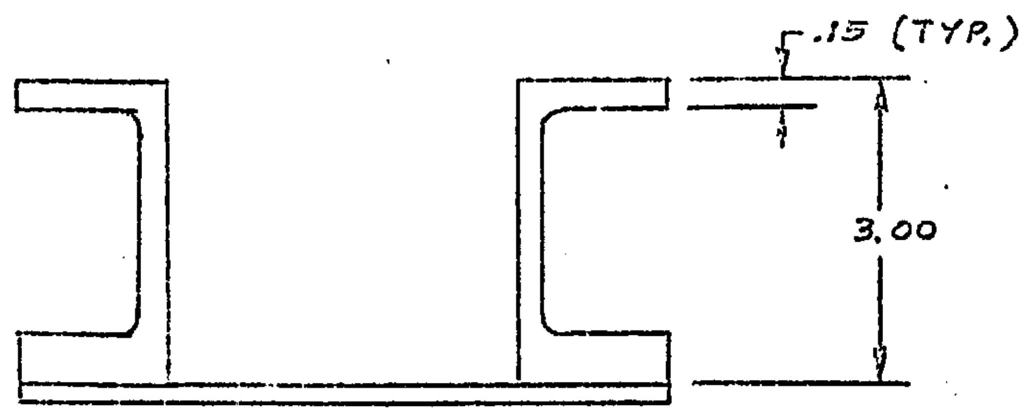


C - C

$AREA = 3.15\ IN.^2$

$I_{YY} = 16.50\ IN.^4$

$I_{ZZ} = 13.70\ IN.^4$



D - D

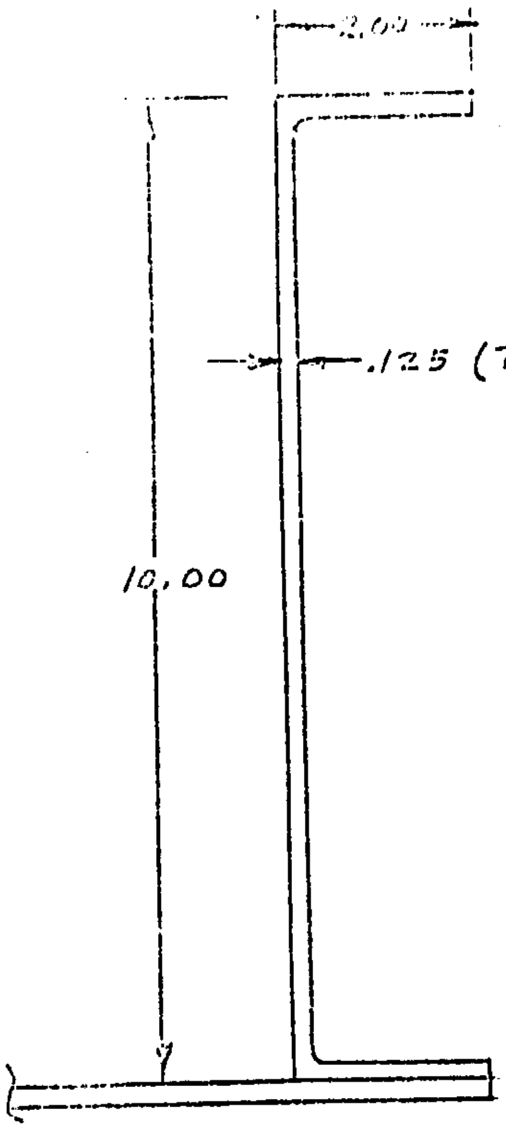
$AREA = 3.24\ IN.^2$

$I_{YY} = 3.08\ IN.^4$

$I_{ZZ} = 14.13\ IN.^4$

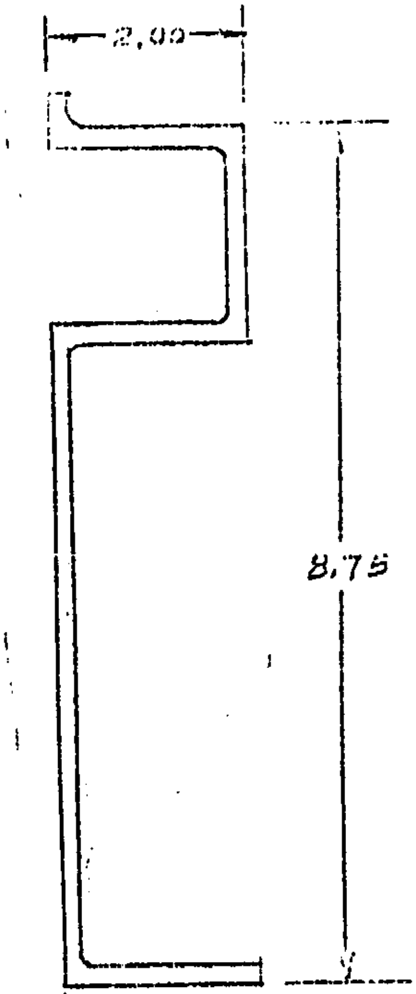
STOL - TRAILING EDGE
CENTER FLAP

CENTER FLAP



A = 1.718 IN.^{2}
I = 15.59 IN.^{4}

H - H



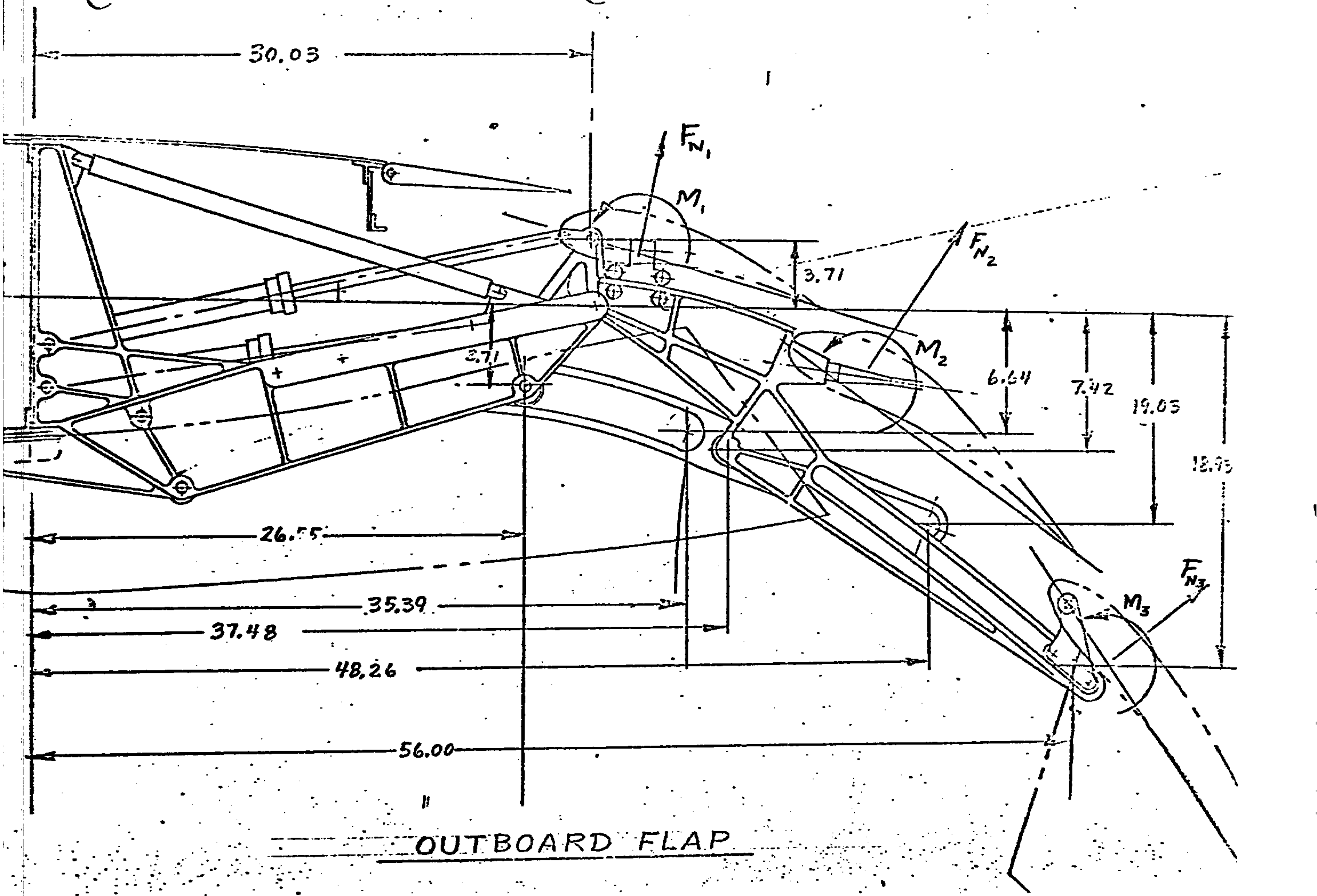
A = 3.56 IN.^{2} (TOTAL)
I = 22.54 IN.^{4} (TOTAL)

J - J

TOTAL A = 4.55 IN.^{2}
TOTAL I = 31.19 IN^4

EACH OF THESE SECTIONS ARE SYMMETRICAL WITH RESPECT TO A VERTICAL ℄.

STOL - TRAILING EDGE



OUTBOARD FLAP

[illegible]TBOARD FLAP

V_1 (TOTAL LOAD) — M (TOTAL MOMENT)

1939	FLAP NO. 3	8330 IN-LB. (LIMIT)
4177	FLAP NO. 2	20,170
8219	FLAP NO. 1	11,547

TAKING MOMENTS ABOUT THE PIVOT POINT ON FLAP NO. 1

$$\Sigma M = 0$$

$$7.45\, R_{B_V} + H_1 1610 \sin 75^\circ + H_2 2083 \sin 55^\circ + H_3 \times 969 \sin 35^\circ = 0$$

$$7.45\, R_{B_V} + 2.03 \times 1555 + 14.31 \times 1710 + 28.36 \times 558 = 0$$

$$7.45\, R_{B_V} + 3156 + 25325 + 15824 = 0$$

$$R_{B_V} = \frac{44307}{7.45} = 5947$$

$$11.13\, R_{B_H} + V_1 1610 \cos 75^\circ + V_2 2083 \cos 55^\circ + V_3 \cdot 969 \cos 35^\circ = 0$$

$$11.13\, R_{B_H} + 2.02 \times 417 + 3.55 \times 1430 + 22.74 \times 795 = 0$$

$$11.13\, R_{B_H} + 842 + 12,654 + 18078 = 0$$

$$R_{B_H} = \frac{31574}{11.13} = 2836$$

$$R_{B_C} = 6600$$

$$11.13\, R_{B_M} + 4330 + 10,095 + 5787$$

$$R_{B_M} = 1,790$$

$$R_B = 6600 + 1790 = 8390 \text{ LB. (LIMIT)}$$

STOL - TRAILING EDGE
OUTBOARD FLAP

FLAP NO. 1 (OUTBOARD)

$w = 46.3$ LB./IN.

$a = 14.44$ IN. $L = $ [illegible] IN.

$t = .020$ IN. $E = $ [illegible] PSI

$$M_x = -\frac{wx^2}{2} + \frac{wLx}{2} - \frac{wa^2}{2}$$

$$M_{x=0} = -\frac{wa^2}{2} = \frac{-46.3}{2}(14.44)^2 = -4816 \text{ IN-LB.}$$

$$f_b = \frac{Mc}{I} = \frac{4816 \times .84}{.60} = \pm 6742 \text{ PSI}$$

$$\frac{q}{t} = \frac{M}{2At} = \frac{3601}{2 \times 27 \times .020} = 3334 \quad \text{PSI}$$

$$y = -.0342 \text{ IN.}$$

FLAP NO. 2 (OUTBOARD)

$w = 60$ LB./IN.

$a = 14.44$ IN. $L = 40.59$

$$M_{x=0} = -\frac{wa^2}{2} = \frac{-60 \times (14.44)^2}{2} = -6255 \text{ IN-LB.}$$

$$f_b = \frac{Mc}{I} = \frac{6255 \times .95}{1.05} = 5659 \text{ PSI}$$

$$\frac{q}{t} = \frac{6250}{2 \times 47 \times .020} = 3324 \quad \text{PSI}$$

$$y = -.0260 \text{ IN.}$$

FLAP NO. 3 (OUTBOARD)

$w = 28$ LB/IN.

$$M_{x=0} = -\frac{wa^2}{2} = -\frac{28 \times (19.67)^2}{2} = 5416 \text{ IN-LB.}$$

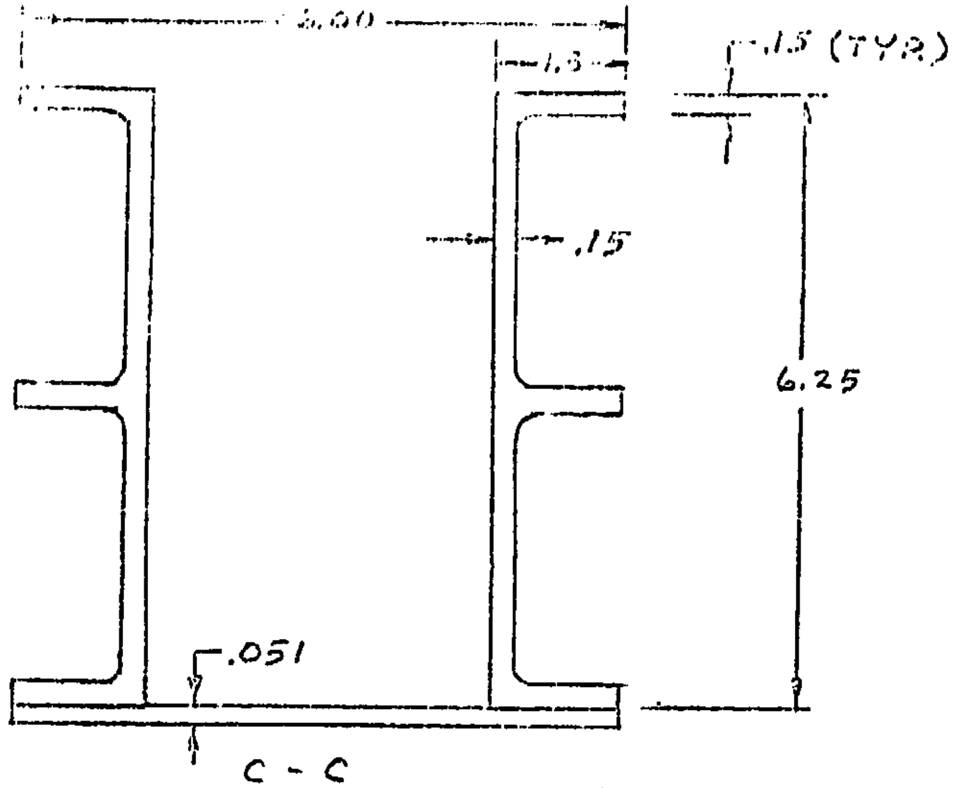
$$f_b = \frac{5416 \times 1.03}{1.15} = 4850 \text{ PSI}$$

$$\frac{q}{t} = \frac{2655}{2 \times 55 \times .020} = 1207 \quad \text{PSI}$$

$$y = -.0113 \text{ IN.}$$

STOL - TRAILING EDGE
OUTBOARD FLAP

OUTBOARD FLAP - BEAM

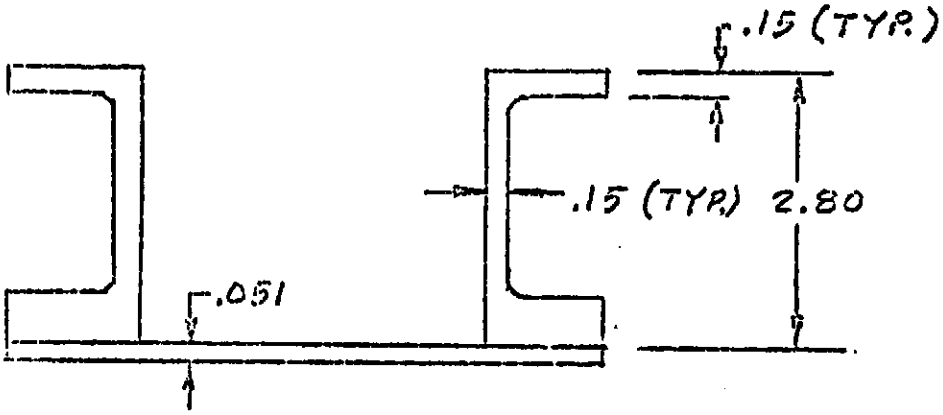


C - C

AREA = 3.25 IN.^{2}

I_{YY} = 12.42 IN.^{4}

I_{ZZ} = 12.39 IN.^{4}



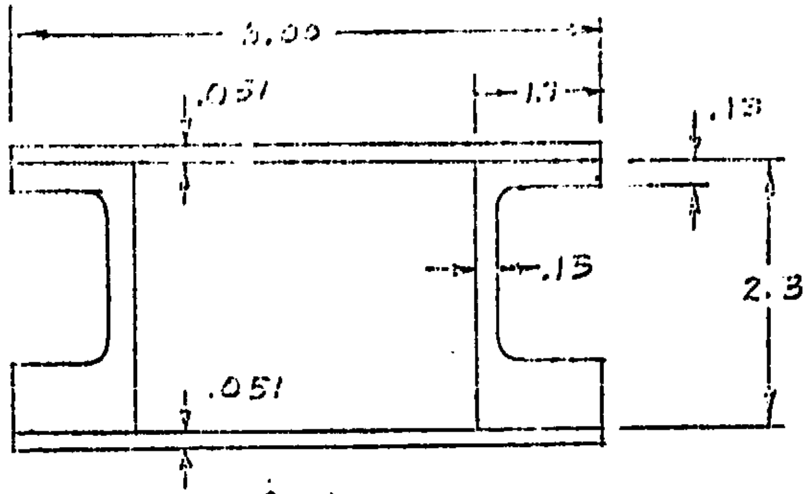
D - D

AREA = 2.98 IN^2

I_{YY} = 2.18 IN.^{4}

I_{ZZ} = 14.16 IN.^{4}

STOL - TRAILING EDGE
OUTBOARD FLAP

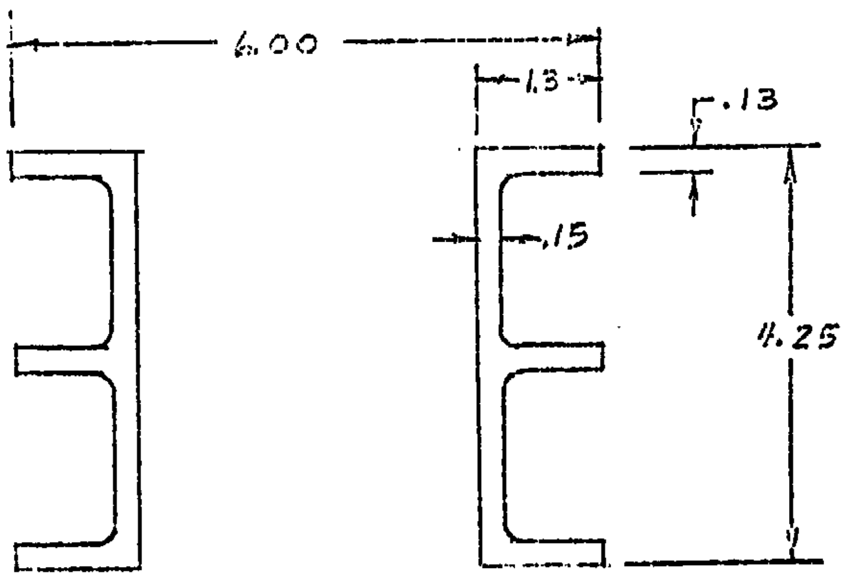


A - A

AREA = 3.02 IN.^{2}

I_{YY} = 1.98 IN.^{4}

I_{ZZ} = 11.70 IN.^{4}



B - B

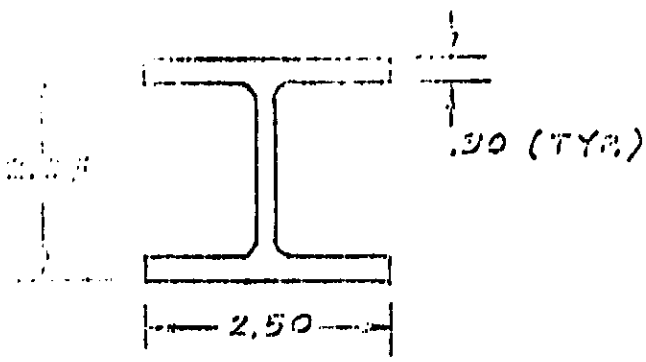
AREA = 2.166 IN^2

I_{YY} = 4.24 IN.^{4}

I_{ZZ} = 8.27 IN.^{4}

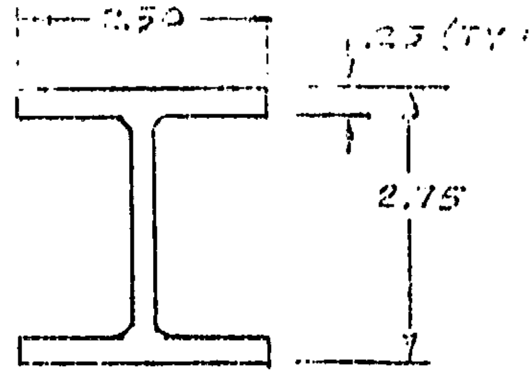
STOL - TRAILING EDGE
OUTBOARD FLAP

BOARD FLAP
TRACK



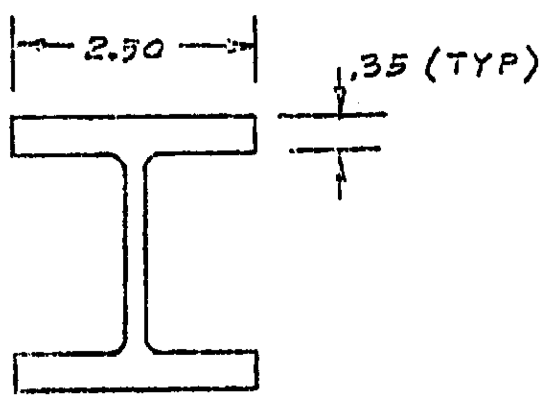
E - E

$A = 1.76\ IN^2$
$I_{YY} = 1.49\ IN.^4$
$I_{ZZ} = .78\ IN.^4$



I - I

$A = 1.59\ IN.^2$
$I_{YY} = 2.09\ IN.^4$
$I_{ZZ} = .65\ IN.^4$

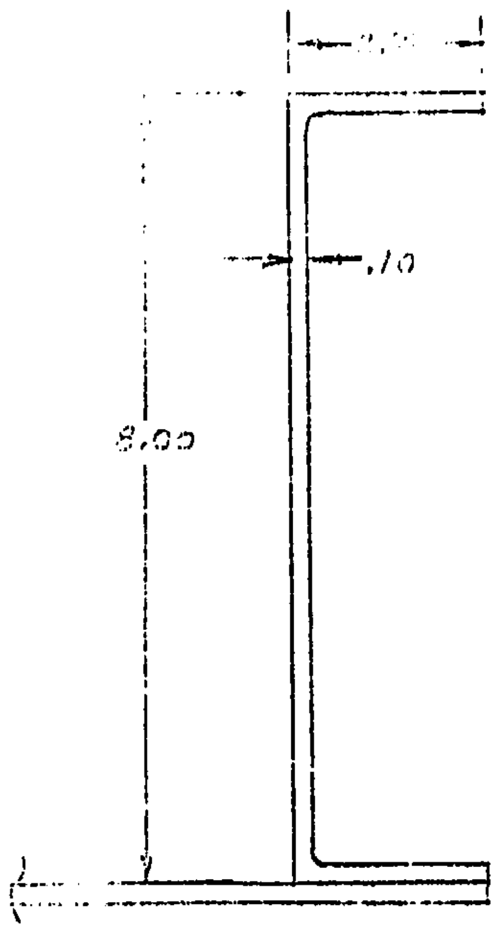


G - G

$A = 2.17\ IN.^2$
$I_{YY} = 2.76\ IN.^4$
$I_{ZZ} = .91\ IN.^4$

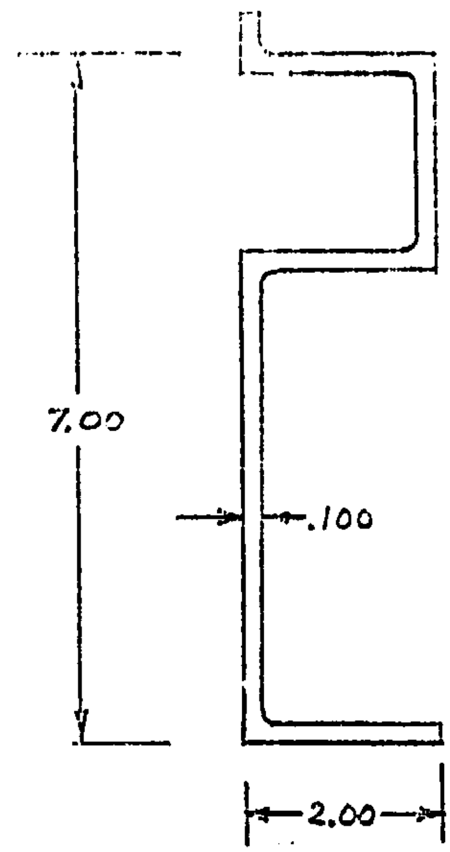
STOL - TRAILING EDGE
OUTBOARD FLAP

OUTBOARD FLAP



H - H

A = 3.24 IN.^{2}
I = 11.72 IN^4
I_{zz} = 2.96

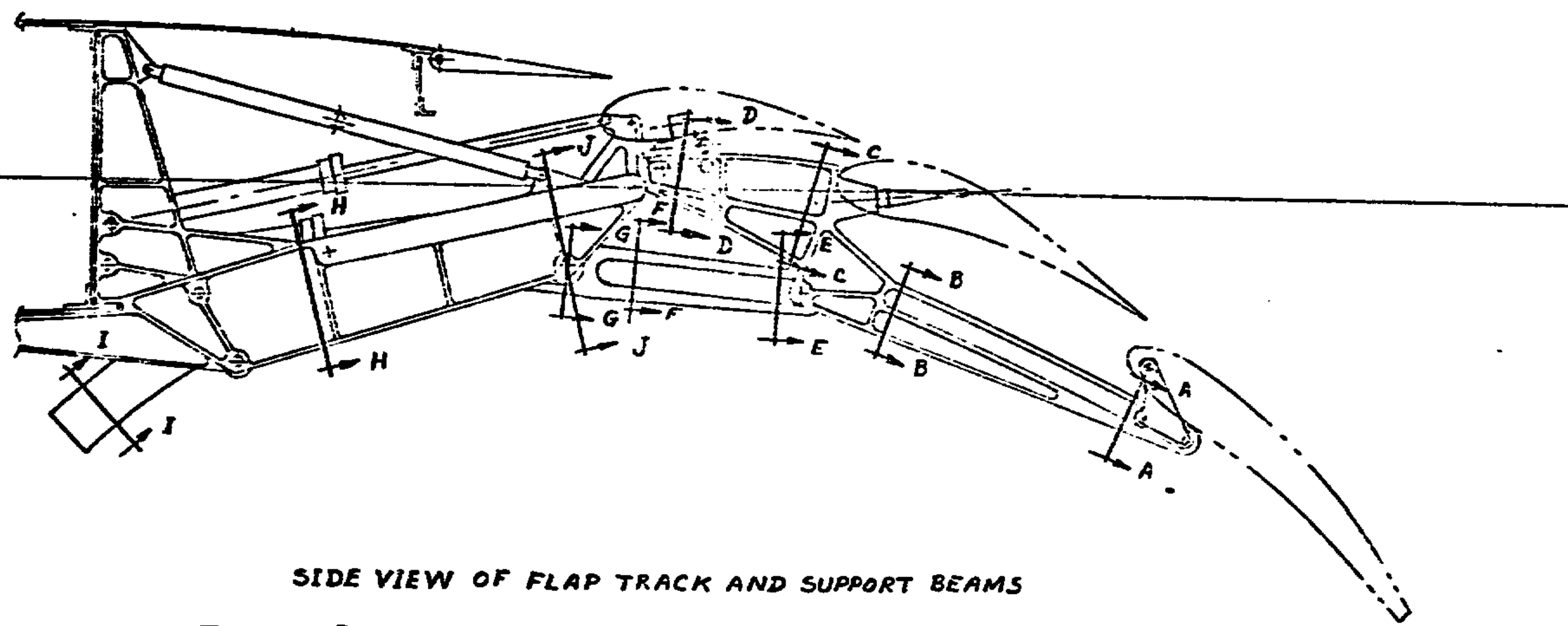


J-J

A = 2.52 IN.^{2}
I = 7.37 IN.^{4}
I_{zz} = .098 IN^4

EACH OF THESE SECTION ARE SYMMETRICAL WITH RESPECT TO A VERTICAL CENTER LINE.

STOL - TRAILING EDGE
OUTBOARD FLAP



SIDE VIEW OF FLAP TRACK AND SUPPORT BEAMS

THE SAME RELATIVE SECTION LOCATIONS ARE USED FOR INBOARD, CENTER, AND OUTBOARD FLAP MECHANISM.
REF. DWG. PD-111-2-010

FIG. II - 15

STOL

SPANWISE DISTRIBUTION OF SECTION HINGE MOMENT (NO BLOWING)

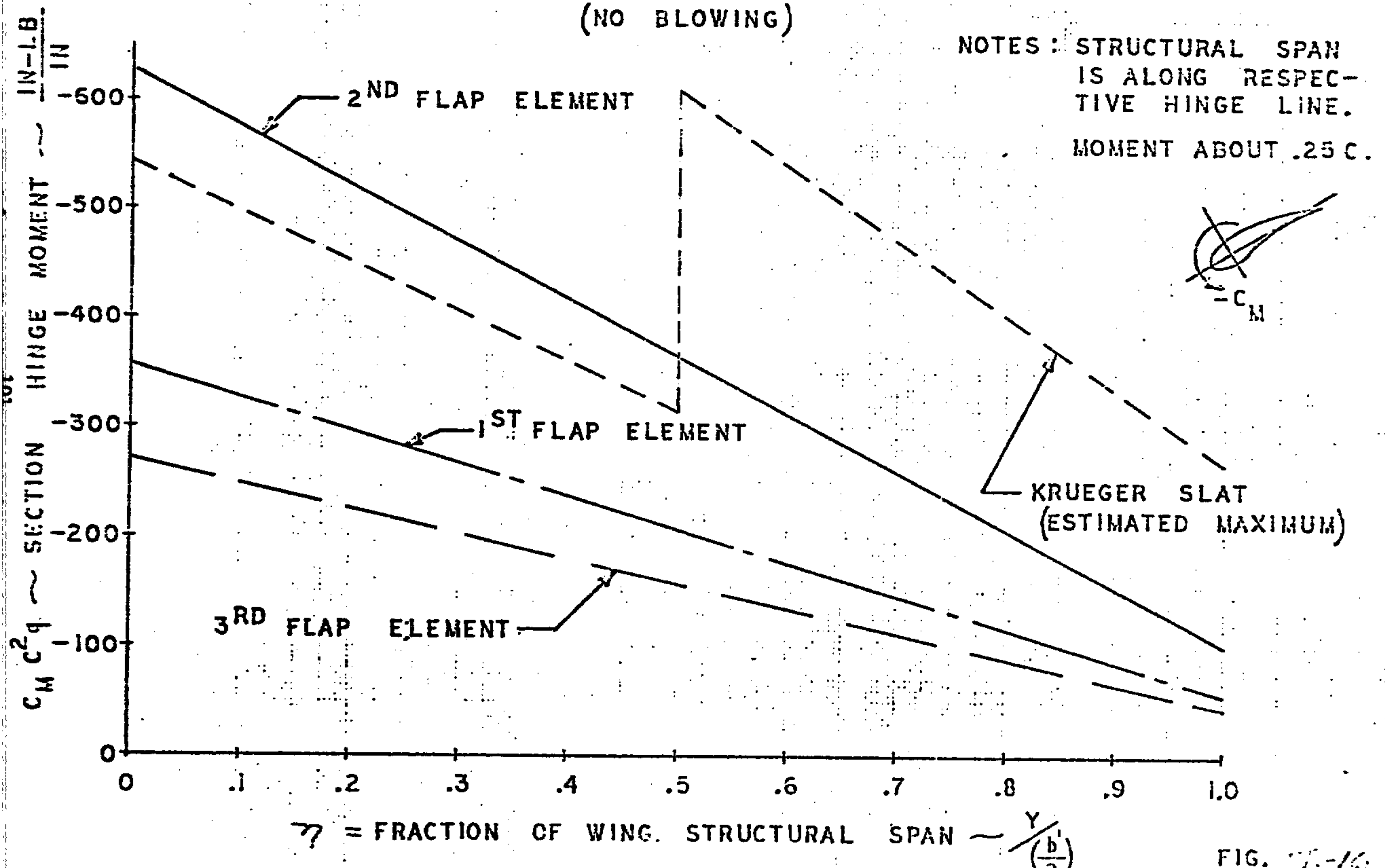
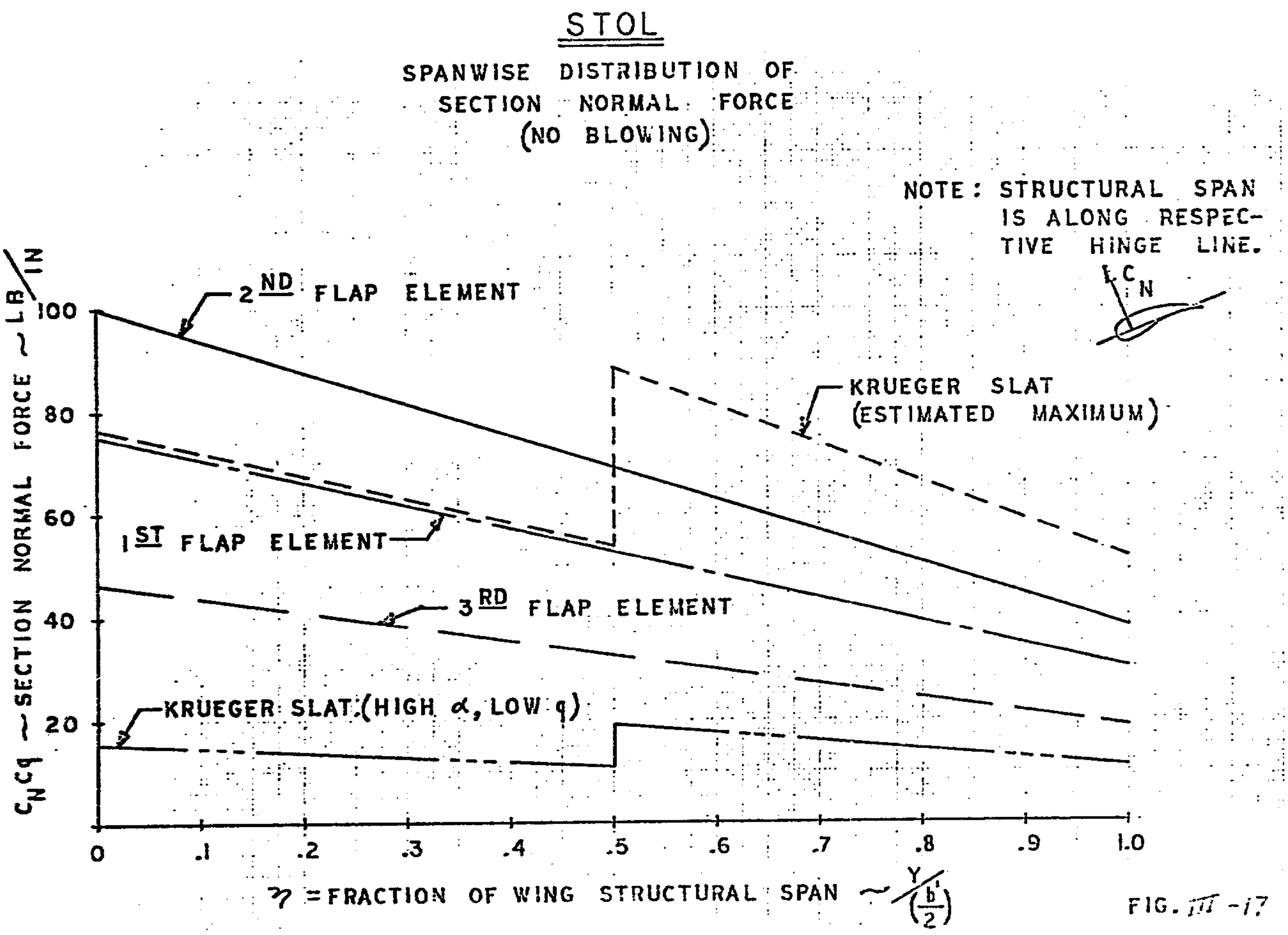


FIG. [illegible]


STOL
SPANWISE DISTRIBUTION OF SECTION NORMAL FORCE (NO BLOWING)
NOTE: STRUCTURAL SPAN IS ALONG RESPECTIVE HINGE LINE.
C_N
2^{ND} FLAP ELEMENT
KRUEGER SLAT (ESTIMATED MAXIMUM)
1^{ST} FLAP ELEMENT
3^{RD} FLAP ELEMENT
KRUEGER SLAT (HIGH α, LOW q)
$C_N c q$ ~ SECTION NORMAL FORCE ~ LB/IN
100
80
60
40
20
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
η = FRACTION OF WING STRUCTURAL SPAN ~ $Y / \left(\frac{b'}{2}\right)$
FIG. III-17

PYLON STRUCTURE

The pylon structure consists of aluminum alloy skin and longerons. The four longerons form the corners of the box beam and torque box. The pylon is internally stiffened with frames and bulkheads. The skin gage is 0.070 inches, except for the bottom panel aft of the thrust mount and side load fitting which is .080 inch.

A criteria was established for the outboard engine nacelle under the following conditions:

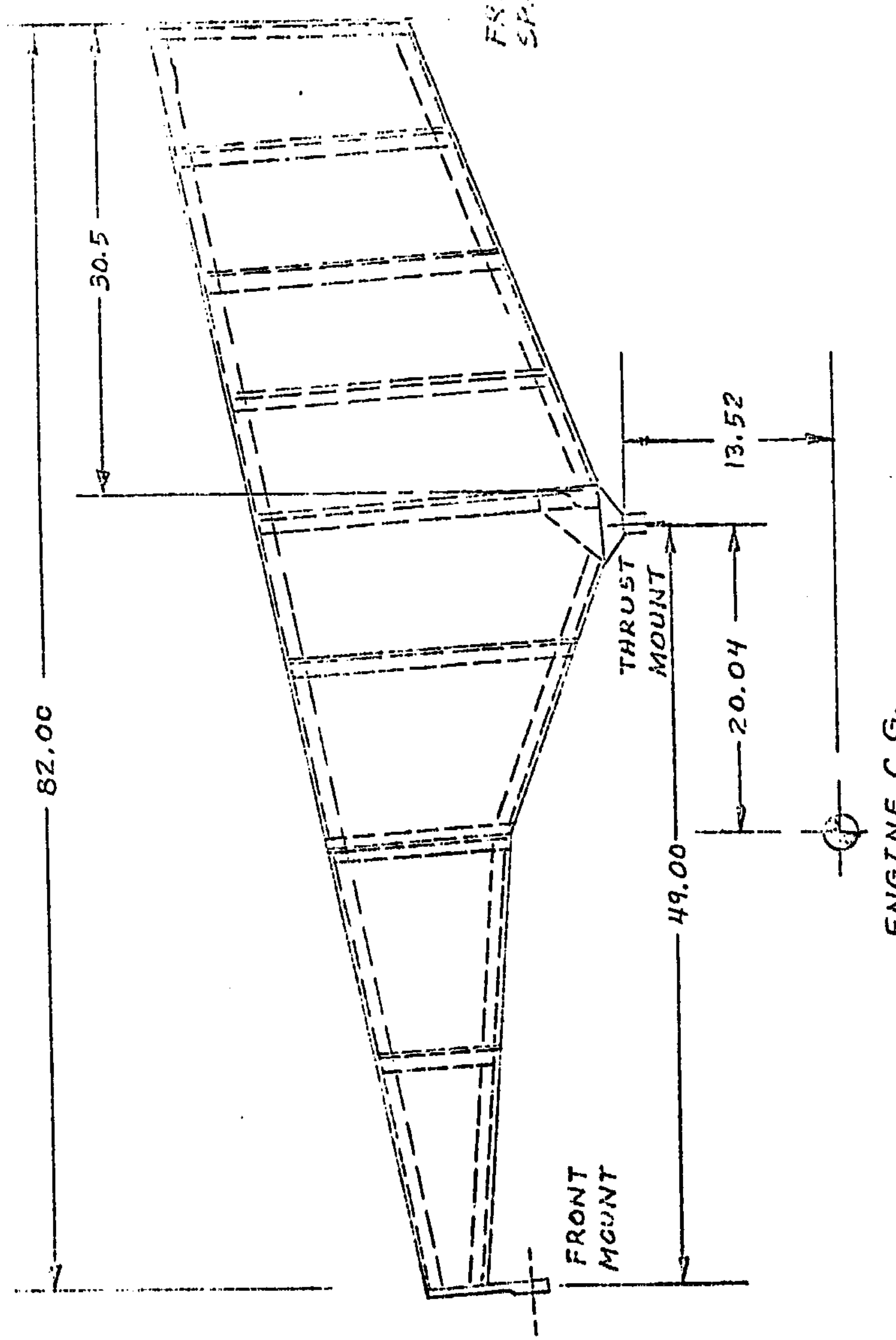
1. Flight pullup with roll and 1.5 cruise thrust.
2. Negative G flight maneuver with 1.5 cruise thrust.
3. Landing condition, positive G.
4. Roll condition, 2.5 side
5. Landing condition, negative G
6. Takeoff condition, negative G, with 1.5 x maximum thrust.
7. Engine seizure.

A summary load sheet is presented showing the distribution of loads applied to the pylon at the front, thrust, vertical, and side mounts.

Three critical conditions design the pylon structure:

1. Landing condition, positive G, designs the lower longerons.
2. Takeoff condition, negative G, with 1.5 times maximum thrust designs the upper longerons.
3. Engine seizure condition designs the shear panels aft of the thrust mount.

STOL - ENGINE PYLON STRUCTURE



STOL - ENGINE PYLON STRUCTURE

SUMMARY OF ENGINE LOADS ON PYLON

STOL
TF-[illegible] ENGINE
OUTBOARD NACELLE - ULTIMATE DESIGN LOADS

CONDITION	FRONT MOUNT		THRUST MOUNT		REAR LEFT		REAR RIGHT	
	VERT.	SIDE	AXIAL	SIDE	SIDE	VERT.	SIDE	VERT.
FLIGHT, 5.25V 1.5 T_c	2038	0	10,125	0	1287	2987	-1287	2987
FLIGHT, -3.0V 1.5 T_c	-5187	0	10,125	0	323	555	323	555
LANDING 6.00V	5256	0	0	0	2574	3414	-2574	3414
ROLL, 2.5S	0	2060		2980	0	-10990	0	10026
LANDING -2.5V	-2190	0	0	0	-1072	-1422	1072	-1422
TAKEOFF, -1.5V 1.5 TM	-4771	0	13950	0	-900	874	900	874
ROLL, 2.5S + 1.5 T_c	-2561	2060	10125	2980		-11954		10026
ENGINE SEIZURE	315,000 IN-LB.							

T_c = 6750 LB. (LIMIT) ENGINE THRUST - POSITIVE FORWARD
T_M = 9100 LB. (LIMIT) MAX. ENGINE THRUST - POSITIVE FORWARD
V = 2015 LB. (LIMIT) ONE FACTOR VERTICAL LOAD - POSITIVE DOWN
S = 2015 LB. (LIMIT) ONE FACTOR SIDE LOAD - POSITIVE INBOARD
LOADS GIVEN AS A GROUP HAVE BEEN COMBINED.

1. VERTICAL FLIGHT LOAD FACTORS ARE BASED ON COMBINATIONS OF THE DESIGN SYMMETRICAL MANEUVER LOAD FACTOR (3.75 G, ULT.) AND LOAD FACTORS RESULTING FROM AN ASSUMED ROLLING ACCELERATION OF 1.75 RAD./SEC.^{2}
2. ALL OTHER FACTORS ARE BASED ON BOEING 707 DESIGN CRITERIA.

TABLE III-7 STOL - ENGINE PYLON LOADS

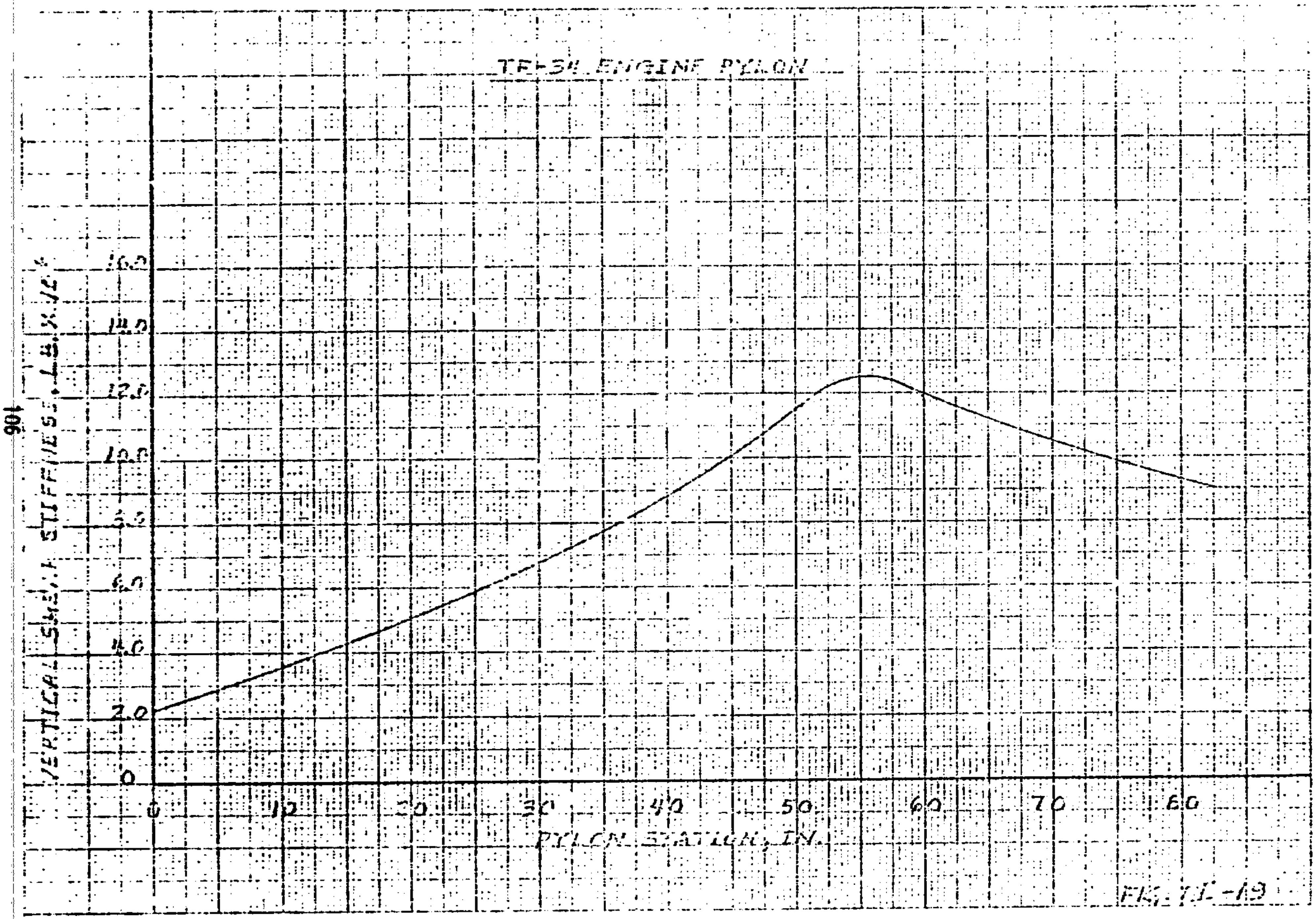

TF-34 ENGINE PYLON
VERTICAL SHELL STIFFNESS
0
2.0
4.0
6.0
8.0
10.0
12.0
14.0
16.0
0
10
20
30
40
50
60
70
80
PYLON STATION, IN.

FIG. II-19

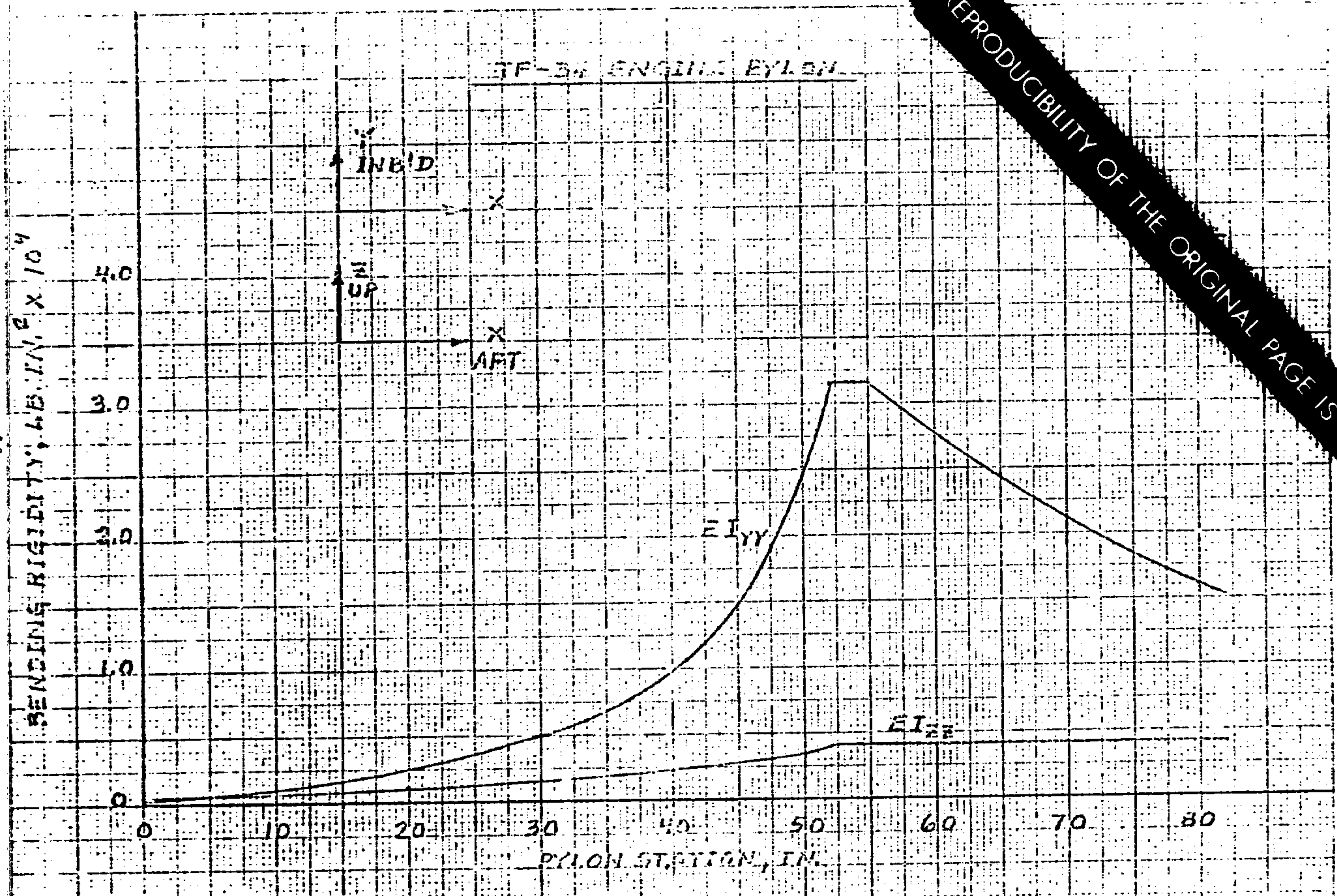

TF-34 ENGINE PYLON
Y INB'D
X
Z UP
X AFT
BENDING RIGIDITY, LB.IN.^{2} X 10^4
4.0
3.0
2.0
1.0
0
EI_{YY}
EI_{ZZ}
0
10
20
30
40
50
60
70
80
PYLON STATION, IN.

FIG. 16-30

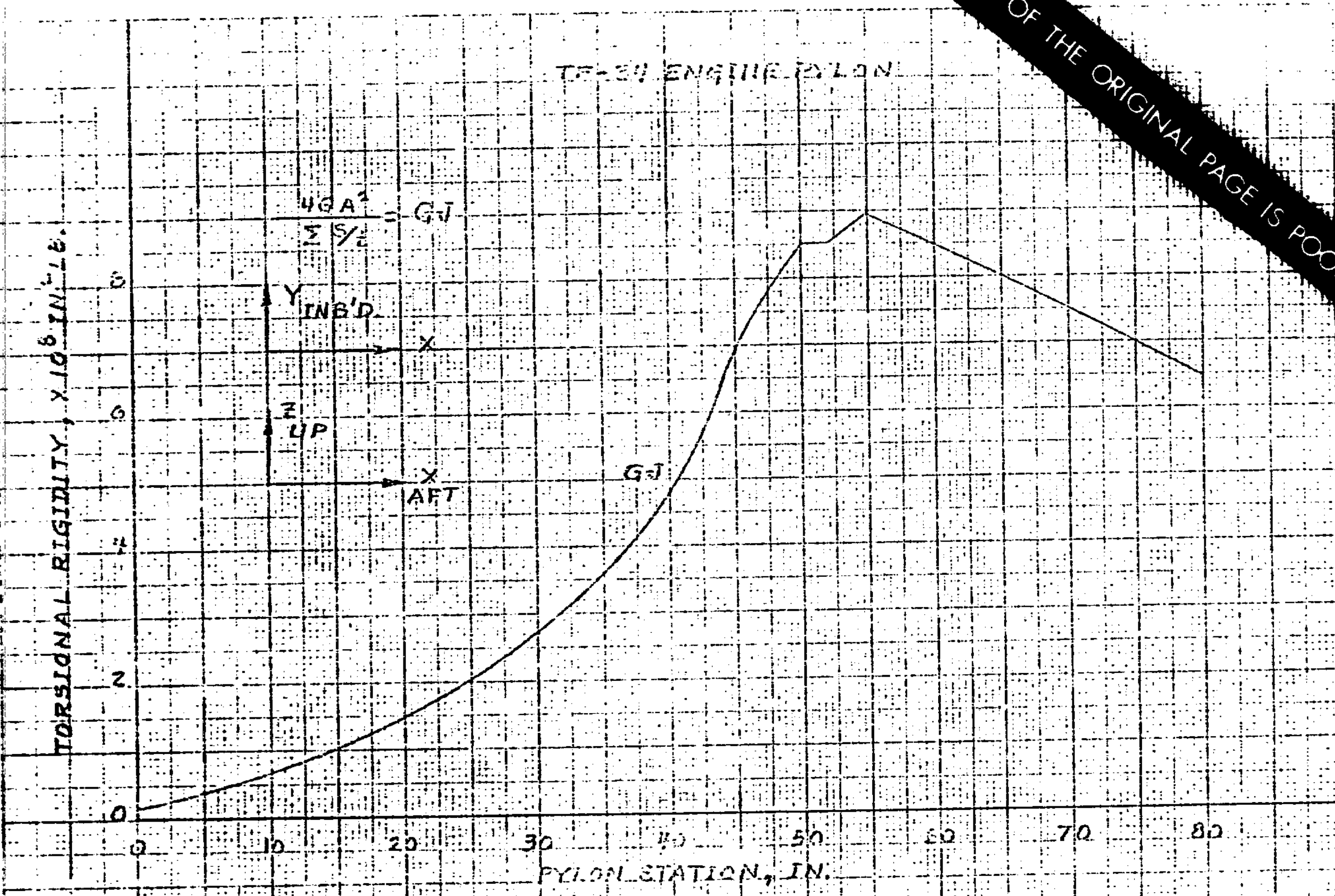
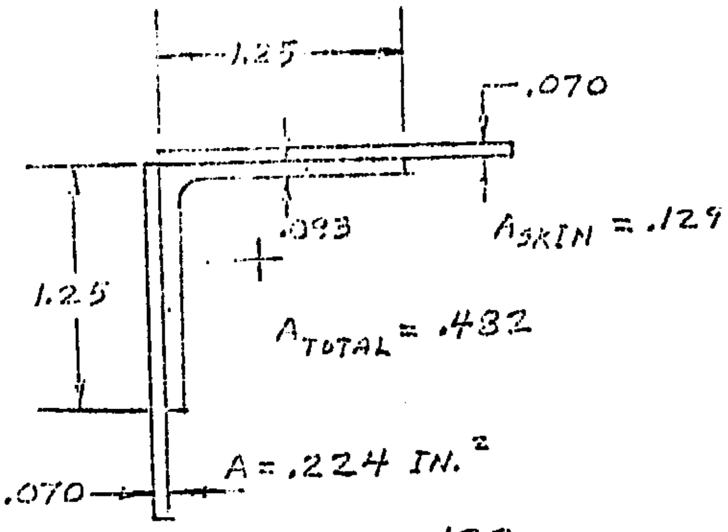

TF-34 ENGINE PYLON
$\frac{4GA^2}{\Sigma S/t} = GJ$
Y INB'D
X
Z UP
X AFT
GJ
0
2
4
6
8
0
10
20
30
40
50
60
70
80
PYLON STATION, IN.
TORSIONAL RIGIDITY, X 10^8 IN.^2-LB.

FIG. III-21

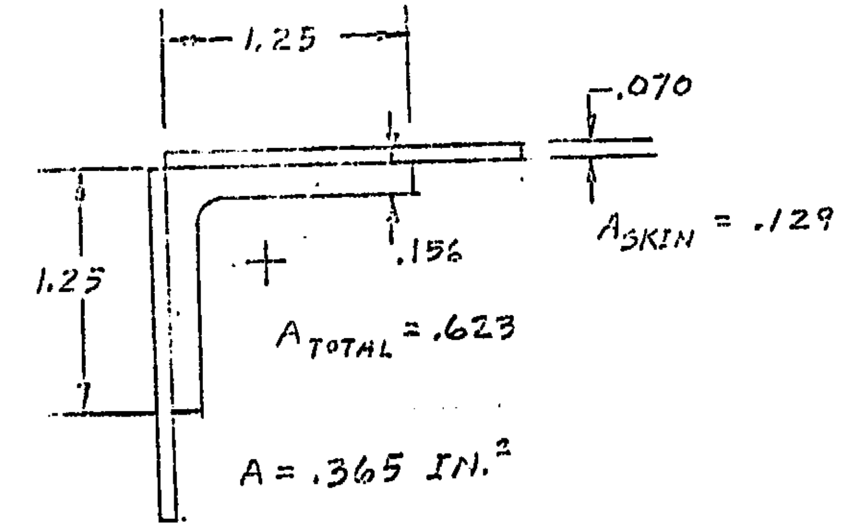


$A = .224\ IN.^2$

$A_{SKIN} = .129$

$$\frac{A}{\Sigma t^2} = \frac{.224}{.0173} = 12.95$$

$F_{cc} = 46.0$ KSI (BAC. DM. 75.231)



$A = .365\ IN.^2$

$A_{SKIN} = .129$

.070

$$\frac{A}{\Sigma t^2} = \frac{.365}{.0486} = 7.5$$

$F_{cc} = 66.0$ KSI (BAC. DM. 75.231)

STOL - ENGINE PYLON STRUCTURE

PYLON
SHEAR PANEL

CHECK THE SHEAR PANELS FOR THE ENGINE SEIZURE CONDITION

ENGINE SEIZURE TORQUE = 315,000 IN-LB. (ULT.)

$$q = \frac{T}{2A} = \frac{315{,}000}{2 \times 230} = 684 \text{ LB./IN.}$$

THE SHEAR STRESS

$$f_s = \frac{q}{t} = \frac{684}{.070} = 9770 \text{ PSI}$$

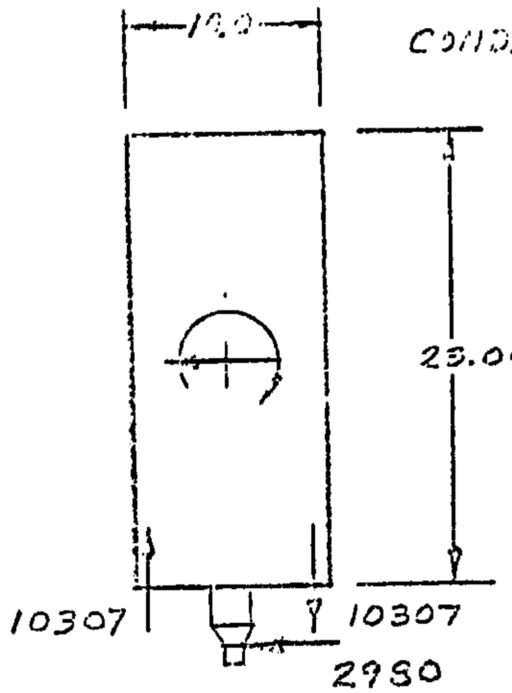
THE ALLOWABLE STRESS

$$F_{s_{CR}} = KE\left(\frac{t}{b}\right)^2 = 7.10 \times 10.3 \times 10^6 \times \left(\frac{.070}{5.0}\right)^2 = 14{,}300 \text{ PSI}$$

PYLON
SHEAR PANELS

CHECK THE SHEAR PANELS AFT OF THE THRUST MOUNT AND
[illegible] FITTING

CONDITION: ROLL, 2.5 G



AREA ENCLOSED = 230 IN.^{2}

TRANSFER THE 2980 LB. LOAD TO THE CENTER OF THE FRAME AS A SHEAR AND A MOMENT

$$T = 14.5 \times 2980 + 10307 \times 10 = 146{,}230 \text{ IN-LB.}$$

$$q_1 = \frac{T}{2A} = \frac{146{,}290}{2 \times 230} = 318 \text{ LB./IN.}$$

$$q_2 = \frac{2980}{2 \times 10} = 154 \text{ LB/IN.} \quad \text{(SIDE LOAD - SHEAR FLOW)}$$

$$\frac{q_1}{t} = \frac{321}{.070} = 4543 \text{ PSI.} \quad \text{(STRESS IN VERTICAL PANELS)}$$

$$F_{s_{CR}} = 7.1 \times 10.3 \times 10^6 \times \left(\frac{.070}{5.0}\right)^2 = 14300 \text{ PSI} \quad \text{(VERTICAL PANE-}$$

THE BOTTOM PANEL IS LOADED BY q_1 AND q_2

$$\frac{q_1 + q_2}{t} = \frac{472}{.070} = 6740 \text{ PSI} \quad \text{(STRESS IN BOTTOM PANEL)}$$

THE ALLOWABLE STRESS IN THE BOTTOM PANEL

$$F_{s_{CR}} = 8.0 \times 10.3 \times 10^6 \times \left(\frac{.070}{5.0}\right)^2 = 16150 \text{ PSI}$$

PANELS ARE ASSUMED MIDWAY BETWEEN CLAMPED AND SIMPLY SUPPORTED EDGES

PYLON
SHEAR PANELS

[illegible] SHEAR PANELS FORWARD OF FRONT MOUNT SPAR.
CONDITION: ROLL, 2.5 G

$$q = \frac{T}{2A} = \frac{105{,}230}{2 \times 130.} = 405 \text{ LB./IN.}$$

$$\frac{q}{t} = \frac{405}{.070} = 5785 \text{ PSI} \quad (\text{VERTICAL PANEL})$$

$$F_{s_{CR}} = 7.2 \times 10.3 \times 10^{7} \times \left(\frac{.070}{5.0}\right)^2 = 14530 \text{ PSI} \quad (\text{VERTICAL PANEL})$$

$$\frac{405 + 154}{.070} = 7985 \text{ PSI} \quad (\text{BOTTOM PANEL})$$

$$F_{s_{CR}} = 16150 \text{ PSI} \quad (\text{BOTTOM PANEL})$$

PANELS ARE ASSUMED MIDWAY BETWEEN CLAMPED AND SIMPLY SUPPORTED EDGES.

THE SHEAR STRESS FOR THE ENGINE SEIZURE CONDITION INCREASES TO

$$\frac{q}{t} = \frac{T}{2At} = \frac{315{,}000}{2 \times 180 \times .070} = 12{,}500 \text{ PSI}$$

BY COMPARISON WITH THE PREVIOUS CHECK FOR THE ENGINE SEIZURE CONDITION IT CAN BE SEEN THAT THE STRUCTURE IS ADEQUATE.

[illegible]ON
SHEAR PANELS
CONDITION: FLIGHT, -3Y, 1.5 T_2,
CHECK SHEAR PANEL FORWARD OF THRUST POINT
TOTAL VERTICAL SHEAR = 4938 LB.

$$q = \frac{V}{2h} = \frac{4938}{2 \times 20} = 123 \text{ LB/IN.}$$

LET $t = .070$

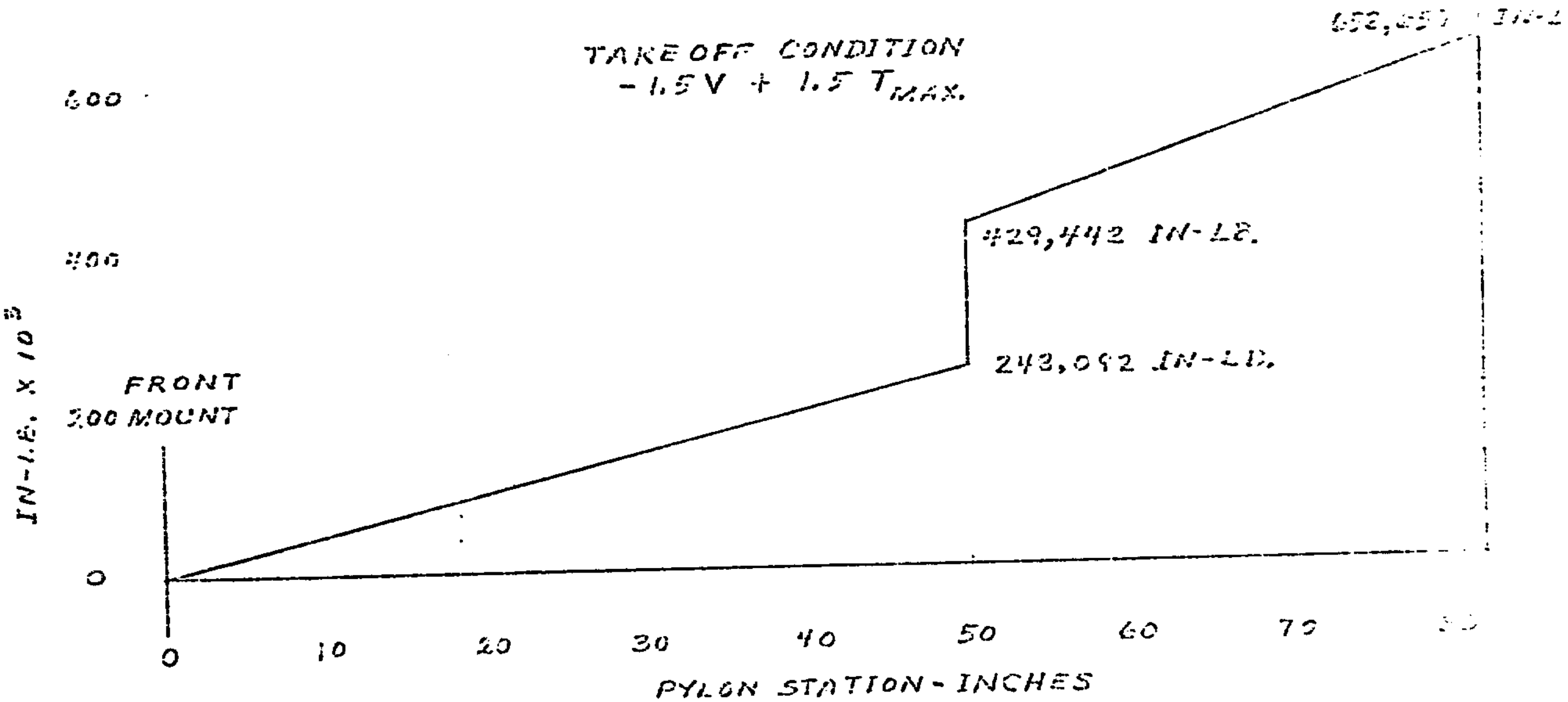
$$\frac{q}{t} = \frac{123}{.070} = 1760 \text{ PSI (VERTICAL PANEL)}$$

$$F_{CR} = 5.7 \times E \times \left(\frac{t}{b}\right)^2 = 5.7 \times 10.0 \times 10^6 \times \left(\frac{.07}{10}\right)^2 = 2790 \text{ PSI}$$

$$\frac{a}{b} = \frac{20}{10} = 2$$

$$K_s = 5.7$$

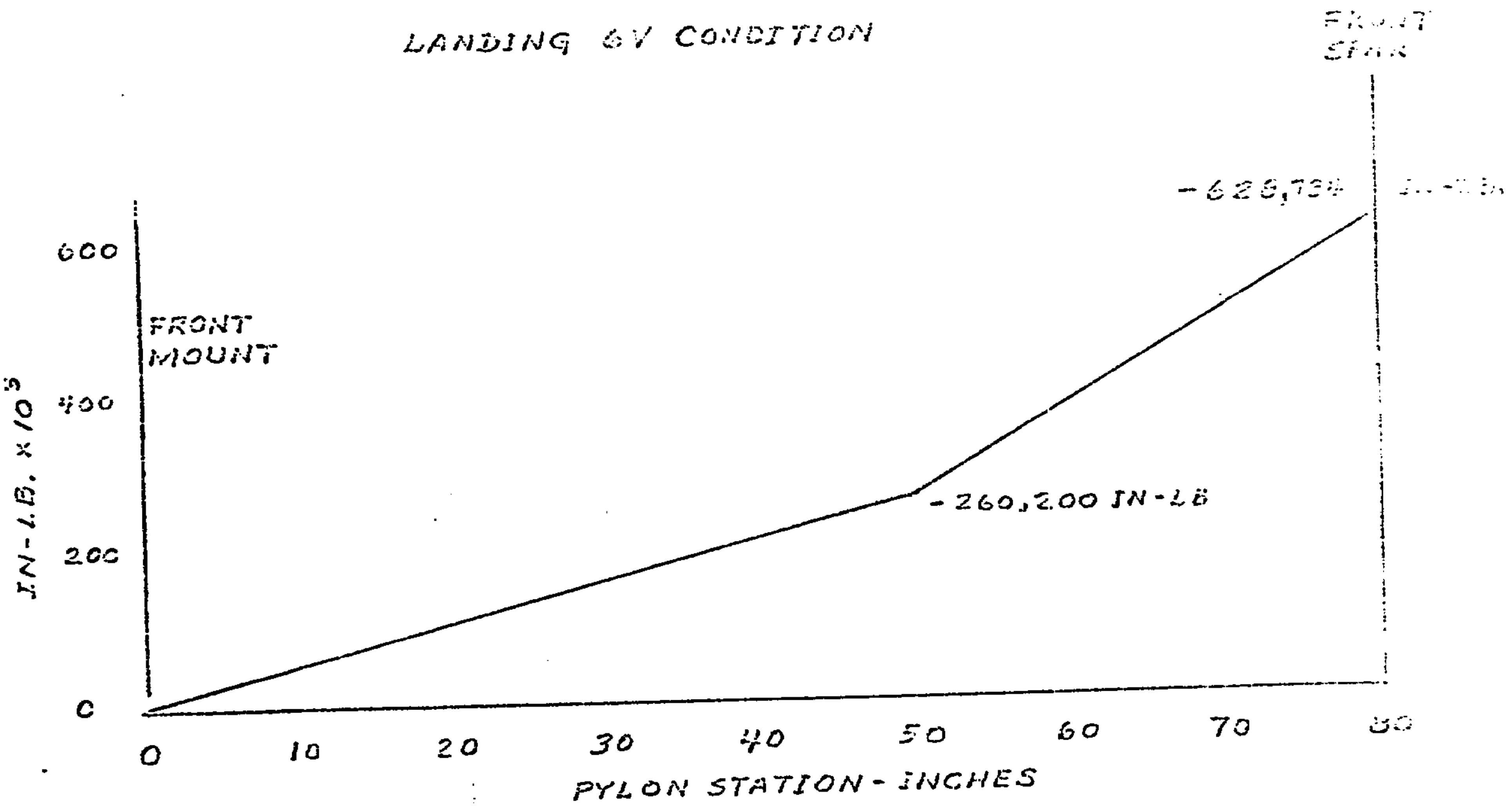
TF-34 ENGINE PYLON
BENDING MOMENT VS. PYLON STATION



POSITIVE BENDING MOMENT PRODUCES COMPRESSION ON UPPER LONGERONS.

TF-34 ENGINE PYLON
BENDING MOMENT VS PYLON STATION

LANDING 6V CONDITION



NEGATIVE BENDING MOMENT PRODUCES COMPRESSION ON LOWER LONGERONS.

FIG. II -

TF-34 ENGINE PYLON
BENDING MOMENT VS. PYLON STATION

CONDITION
SYMMETRICAL FLIGHT + ROLL
5.25 V + 1.5 Tc

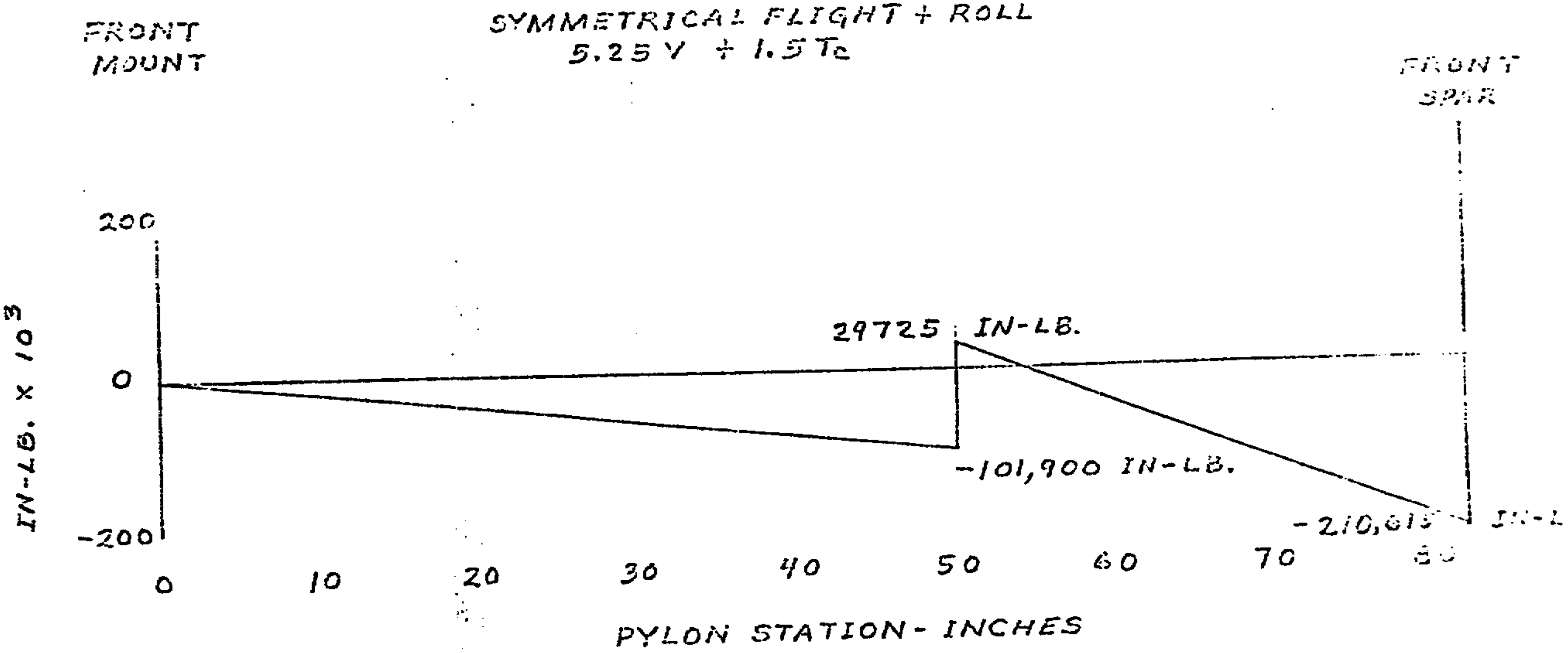


FIG.

Section IV. WEIGHTS

The weight, mass distribution, and moment of inertia data for the STOL wing is based on a combination of structural and deterministic estimating methods.

The following components were evaluated deterministically based on preliminary stress sizing.

Wing Box shell structure

Trailing edge flap support tracks and carriages

The engine and nacelle weight is based on the S-3A actual weight data. All other component weights were obtained from conventional preliminary design statistical weight estimating methods.

The centrodial locations of the wing components are shown on Drawing PD-111-2-009 mass properties layout.

Tables IV-1 through IV-9 are summary tables indicating weights, centroid location and inertias of wing and components.

Figures IV-1 and IV-2 are mass weight distribution plots for the wing box and complete wing per side, respectively.

FIXED LEADING EDGE

STOL

REFERENCE AXIS: X= F.S._____ Y=B.L. 0

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTA LB-IN. [illegible] I_{xo}	I_{yo}	I_{zo}
OUTBOARD							
1	.984—.888	17	218.0	401.0	3.48	.34	3.48
2	.888—.795	18	198.0	361.0	3.55	.51	3.55
3	.795—.669	28	173.0	313.0	9.88	.92	9.88
4	.669—.546	30	146.0	259.0	10.13	1.13	10.13
	Σ	93					
INBOARD							
1	.483—.327	28	98.3	171.0	14.30	3.36	12.58
2	.274—.165	23	56.2	89.0	7.43	2.95	6.06
	Σ	51					

LEADING EDGE FLAPS

STOL

EFERENCE AXIS: X= F.S.____ Y=B.L. O

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTIA LB-IN. [illegible] I_{xo}	I_{yo}	I_{zo}
UTBOARD							
1	.984—.888	17	218.0	401.0	3.48	.34	3.48
2	.888—.795	18	198.0	361.0	3.55	.51	3.55
3	.795—.669	28	173.0	313.0	9.88	.92	9.88
4	.669—.546	30	146.0	259.0	10.13	1.13	10.13
	Σ	93					
INBOARD							
1	.483—.327	28	98.3	171.0	14.30	3.36	12.58
2	.274—.165	23	56.2	89.0	7.43	2.95	6.06
	Σ	51					

WING BOX

STOL

REFERENCE AXIS: X = F.S.____ Y = B.L.O

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTIA LB-IN. [illegible] I_{xo}	I_{yo}	I_{zo}
1	1.0 — .9	53.	237.5	408.	9.87	2.84	12.00
2	.9 — .8	58.	217.5	365.	11.29	4.02	14.31
3	.8 — .7	68.	198.5	322.	13.92	5.93	18.38
4	.7 — .6	83.	178.8	279.	17.90	8.89	27.43
5	.6 — .5	111.	158.7	236.	25.28	14.31	36.03
6	.5 — .4	150.	139.0	193.	36.14	22.91	53.35
7	.4 — .3	192.	119.6	150.	49.01	34.31	74.78
8	.3 — .2	235.	100.2	107.	63.61	48.55	100.08
9	.2 — .1	270.	79.7	64.	78.06	64.79	126.75
10	.1 — .0	282.	78.7	21.7	88.51	71.81	135.71
	Σ	1502.					

WING FIXED TRAILING EDGE

STOL

(RE)FERENCE AXIS: X= F.S.______ Y=B.L.0

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTIA LB-IN.		
					I_{xo}	I_{yo}	I_{zo}
1	1.0 — .9	9	253.5	411.3	1.660	.254	1.660
2	.9 — .8	13	237.0	367.0	2.518	.487	2.518
3	.8 — .7	17	222.0	324.0	3.473	.819	3.473
4	.7 — .6	21	203.0	281.0	4.545	1.264	4.545
5	.6 — .5	26	185.8	232.0	6.359	2.058	6.359
6	.5 — .4	30	168.8	194.0	7.333	2.645	7.333
7	.4 — .3	34	151.8	150.2	8.854	3.541	8.854
8	.3 — .2	38	134.8	107.8	10.553	4.616	10.533
9	.2 — .138	24	121.5	73.4	4.514	3.136	4.514
	Σ	212					

SPOILERS

STOL

(R)EFERENCE AXIS: X = F.S. _____ Y = B.L. 0

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTIA LB-IN. (illegible) I_{xo}	I_{yo}	I_{zo}
1	1.0—.914	9.5	276.0	413.6	1.27	.14	1.40
2	.914—.829	10.3	251.0	376.0	1.38	.15	1.52
3	.829—.744	11.1	239.0	340.0	1.48	.16	1.63
4	.744—.658	11.8	226.5	305.0	1.58	.18	1.74
5	.658—.573	12.6	213.0	269.0	1.69	.19	1.86
6	.573—.488	13.4	199.0	231.0	1.79	.20	1.97
7	.488—.402	14.1	186.0	194.0	1.90	.21	2.08
8	.402—.316	14.9	172.0	156.0	2.00	.22	2.20
9	.316—.231	16.0	157.0	108.0	2.15	.24	2.36
10	.231—.143	16.4	144.0	81.0	2.20	.24	2.42
	Σ	130.1					

TRAILING EDGE FLAPS

STOL

REFERENCE AXIS: X = F.S._____ Y = B.L. 0

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTIA LB-IN. [illegible]		
					I_{xo}	I_{yo}	I_{zo}
OUTBOARD FLAPS							
1	.750–.598	43	225.2	289.0	15.7	.8	16.8
2	.750–.598	58	233.3	289.0	21.3	2.1	23.8
3	.750–.598	65	238.5	289.0	24.0	3.0	27.5
INTERMEDIATE FLAPS							
1	.598–.395	68	198.4	212.8	44.1	1.9	46.4
2	.598–.395	91	208.2	212.8	59.1	4.6	64.5
3	.598–.395	102	214.0	212.8	66.4	6.4	74.0
INBOARD FLAPS							
1	.395–.143	100	163.5	114.0	99.4	3.9	104.0
2	.395–.143	135	175.0	114.0	134.3	9.5	145.5
3	.395–.143	151	182.2	114.0	150.4	13.4	166.0
	Σ	813					

AILERON

STOL

REFERENCE AXIS: X = F.S.______ Y = B.L. 0

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTA LB.-IN. $\otimes$ 10^{-3}		
					I_{XO}	I_{YO}	I_{ZO}
1	1.0 — .750	100	265.3	376.7	97.5	5.1	103.5

DISCRETE MASSES

STOL

REFERENCE AXIS: X = F.S._____ Y = B.L. 0

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTIA LB-IN.[illegible] I_{xo}	I_{yo}	I_{zo}
AILERON	SUPPORT/ACTUATOR						
1	.948	25	263.0	410.7	.5	3.8	3.4
2	.803	30	247.0	348.0	.6	4.6	4.0
L.E. FLAP	ACTUATORS						
OUTBOARD							
1	.926	10.0	218.0	401.0	.341	.741	.667
2	.833	12.5	198.0	361.0	.426	.926	.833
3	.723	15.0	173.0	313.0	.512	1.112	1.000
4	.598	17.5	146.0	259.0	.597	1.297	1.167
INBOARD							
1	.395	20.0	98.3	171.0	.682	1.482	1.333
2	.205	22.5	56.2	89.0	.768	1.668	1.500
ENGINE	PYLONS						
OUTBOARD	.499	145	98	216.0	6.0	81.6	78.0
INBOARD	.290	145	53	125.6	6.0	81.6	78.0

DISCRETE MASSES

STOL

REFERENCE AXIS: X = F.S._____ Y = B.L. 0

ELEMENT NO.	η STA	WEIGHT LBS.	$\overline{X}$ IN.	$\overline{Y}$ IN.	MOMENT OF INERTIA LB-IN. I_{xo}	I_{yo}	I_{zo}
ENGINE OUTBOARD	.499	1870	81	216.0	584.4	4,281.5	4,281.5
ENGINE INBOARD	.290	1870	35.3	125.6	584.4	4,281.5	4,281.5
T.E. FLAP	TRACKS / ACTUATORS						
OUTBOARD							
1	.718	165	237.0	311.0	5.8	68.4	67.7
2	.630	191	224.0	273.0	6.9	84.4	78.4
INTERMEDIATE							
1	.558	237	213.0	241.8	8.3	104.8	97.2
2	.438	263	194.0	189.8	9.2	116.3	107.9
INBOARD							
1	.342	309	180.0	148.2	10.9	136.5	126.7
2	.192	335	158.0	83.2	12.0	148.0	137.6

STOL

WING BOX SPANWISE WT DISTRIBUTION

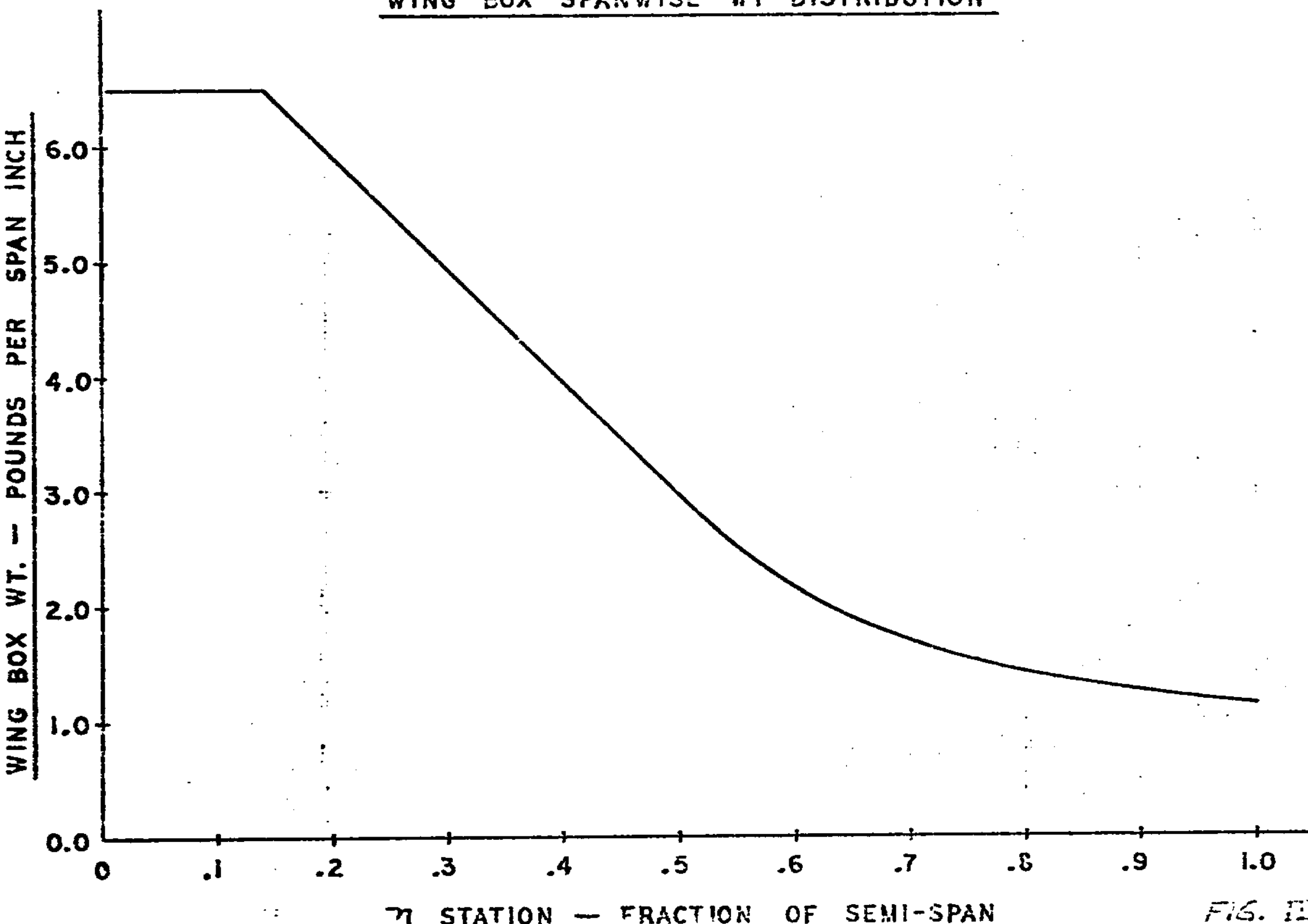


FIG. III-1

STOL

WING WT DISTRIBUTION

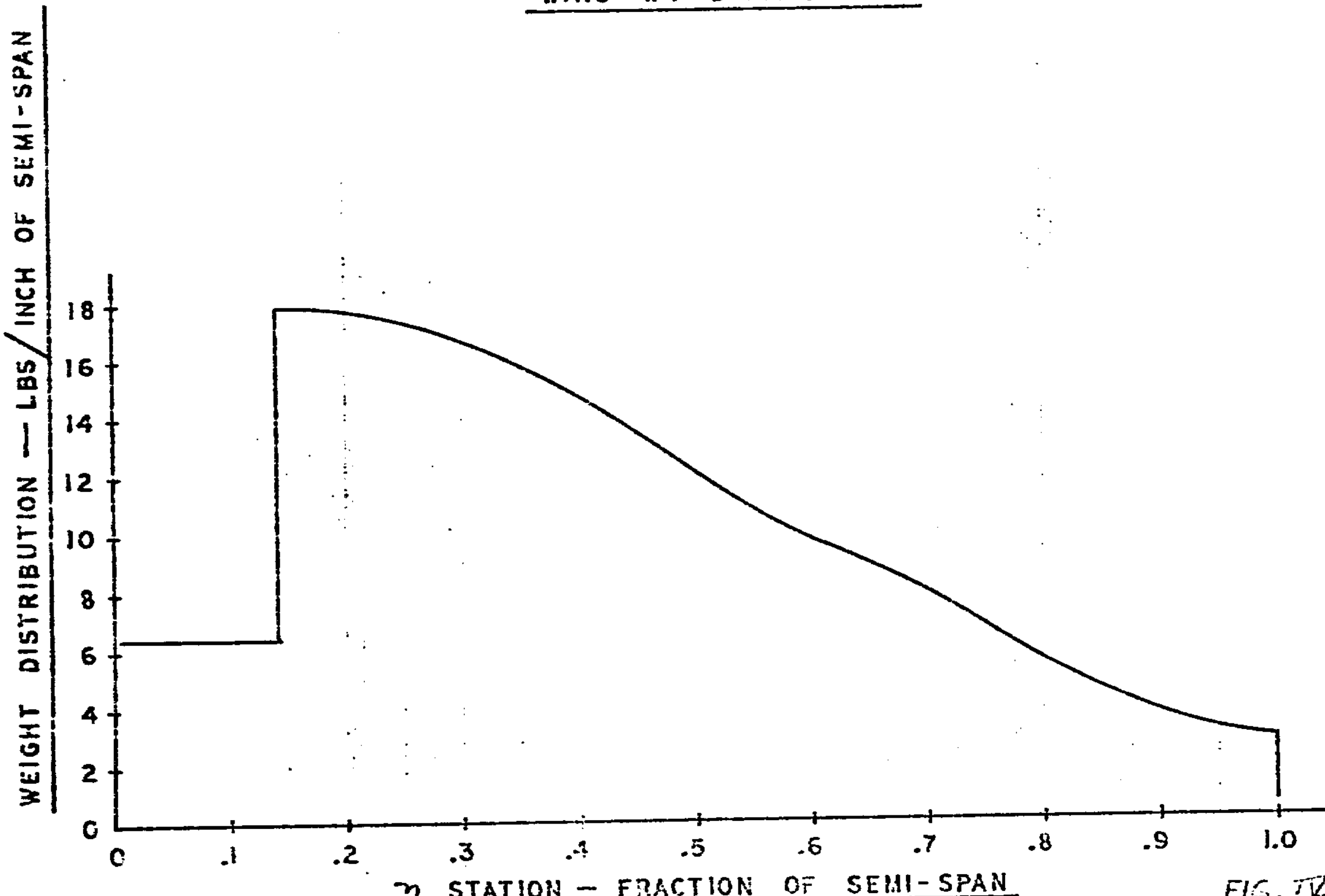
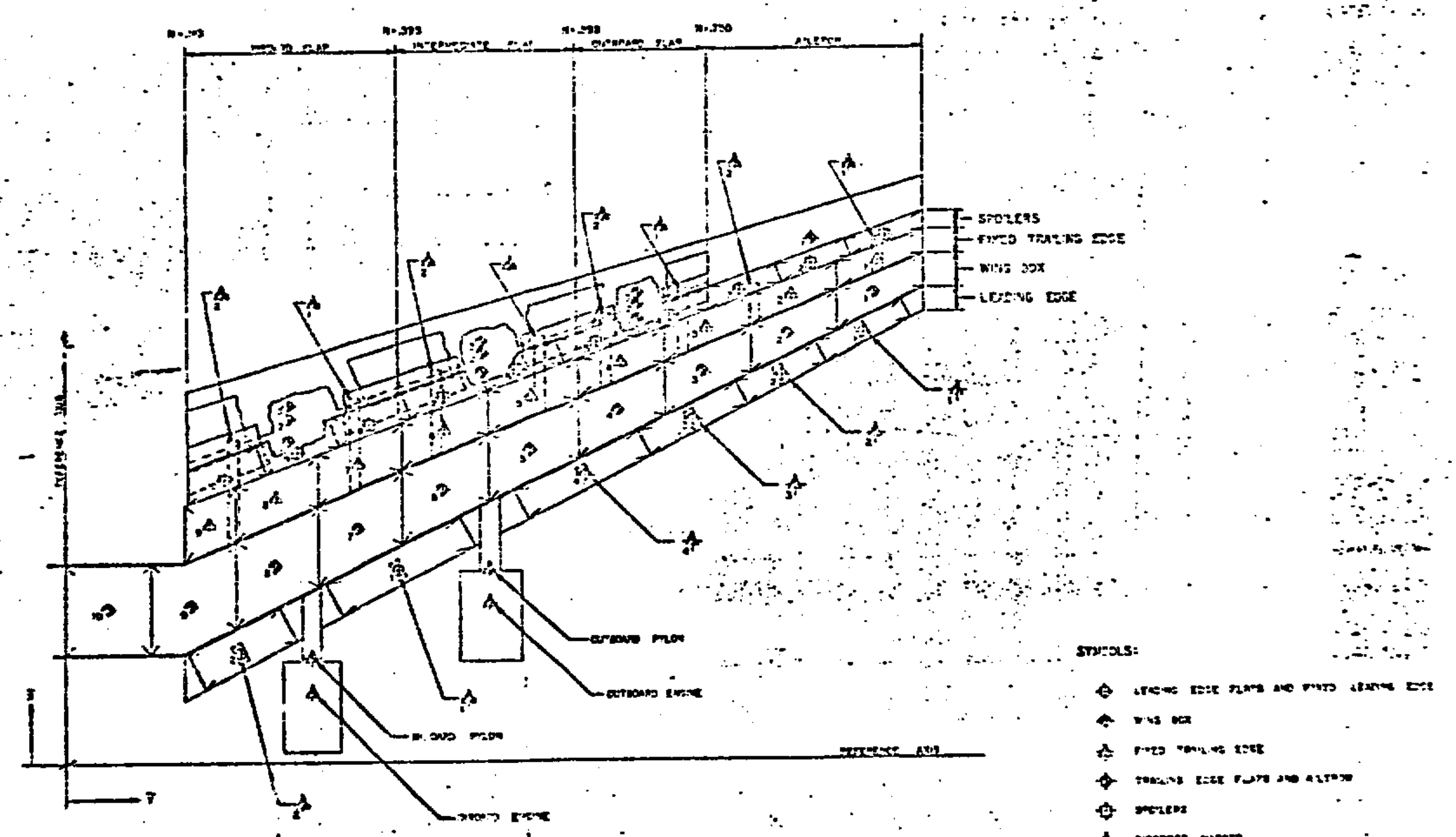


FIG. IV-2



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